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JET PROPULSION LABORATORY
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A PROGRAM ANALYSIS
FOR
LUNAR EXPLORATION

By the Advanced Lunar Studies Team

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FOREWORD

In 1966 NASA asked JPL to begin a continuing study of post-Apollo lunar scientific exploration, to identify priority goals and to help to define needed developments. Initial results of the study were given (Ref. 1) at the January, 1967, NASA Lunar Exploration Program Review. This report presents the conclusions derived by the JPL group through mid-1967. The study has been jointly supported and monitored by Office of Manned Space Flight (OMSF) and Office of Space Sciences and Applications (OSSA) people, and is intended to yield program plans based on efficient combinations of manned and unmanned missions to the Moon. The initial work has been kept relatively unconstrained. In particular, NASA plans (as reflected in the annual OSSA Prospectus, for example) have been treated as general guides rather than as specific requirements to be met by the programs considered. The scientific and technical opinions expressed are those of the JPL authors and are offered as independent judgments based on our analyses and our flight program experience. We do not intend to state policy or give commitments of future effort on the part of the Laboratory; we do present specific recommendations for future effort in areas where the analysis shows work to be needed. The following members of the Advanced Lunar Studies Team have been directly involved in preparing this document:

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I. INTRODUCTION

The purpose of this study is to describe a logical group of goals and then to examine the mission plans for a lunar exploration program directed toward understanding the nature of the Moon. Previous studies such as those of the National Academy of Sciences (Ref. 2, 1965) and the President's Science Advisory Committee (Ref. 3, 1967) have served to outline the major objectives, scope, and pace of a post-Apollo lunar program. In the present study the interrelationships among objectives have been examined for their effect on experimental strategy, and development plans have been studied to identify the combinations of existing and new systems that could be used to carry out the program. In order to emphasize questions of immediate interest, attention has been focused on (a) priority questions in the scientific rationale and (b) near-future engineering developments. Later goals such as exploitation of lunar resources and operation of lunar bases are excluded.

The content of the study can be summarized as follows:

(1) Why--Priority scientific goals, reasons for program.

(2) What--Investigations required to reach goals.

(3) When--Logical sequence of initial measurements, later surveys, and detailed explorations.

(4) Where--Likely locations on Moon.

(5) How--Experiments and carrier vehicles.

These subjects are taken up, in approximately the order listed, in the subsequent Sections of this report.

II. SCIENTIFIC EXPLORATION PROGRAM

Using the Space Science Board's reasoning and the associated 15 questions (Ref. 2) as a basis, the first step in the scientific part of the study was to devise a rationale that would display logical sequences of investigation. This work resulted in the chart shown in Fig. 1. On this chart* there are identified a number of major lines of investigation, each proceeding through characteristic phases ranging downward from initial or gross measurements to later, more refined investigations. The phases are, of course, somewhat related to assumed advances in technology, but they also reflect the intrinsic ordering of any sequential scientific investigation. In the present study we have confined our attention mostly to the early phases, since these are the ones now influencing program decisions, and also since the needs of the later phases are naturally more uncertain at this time.

The next step was to write down sets of measurements that could yield the data required for each phase of each investigation. Of course, some arbitrary choices were needed here, since, for example, we could not know in advance how many measurements of a given phenomenon would be needed for a full understanding of it. The resulting measurement lists were published in Ref. 1.

The next step in the study was to identify priority requirements. Up to this point we had been considering a program dedicated to answering all recognized lunar questions, without any ordering except that imposed by the investigative logic itself. To be useful for program planning, the study should yield a more selective output. Rather than trying to weigh the relative merits of the several scientific disciplines, we elected to consider the questions and measurements in four categories:

(1) Critical Data

(2) Cosmogony

(3) Exobiology

(4) Astrogeology

These categories are arranged in priority from the specific goals of lunar science to more general goals. 'Critical Data' refers to that group of experiements whose results are needed for effective planning of a continuing program. These experiments belong in general to Phase 1 in Fig. 1. For example, an early indication of the presence or absence of seismic activity is essential for planning the lunar seismological program.

*Fig. 1 is a slightly revised version of the chart presented in January as Fig. 9 of Ref.

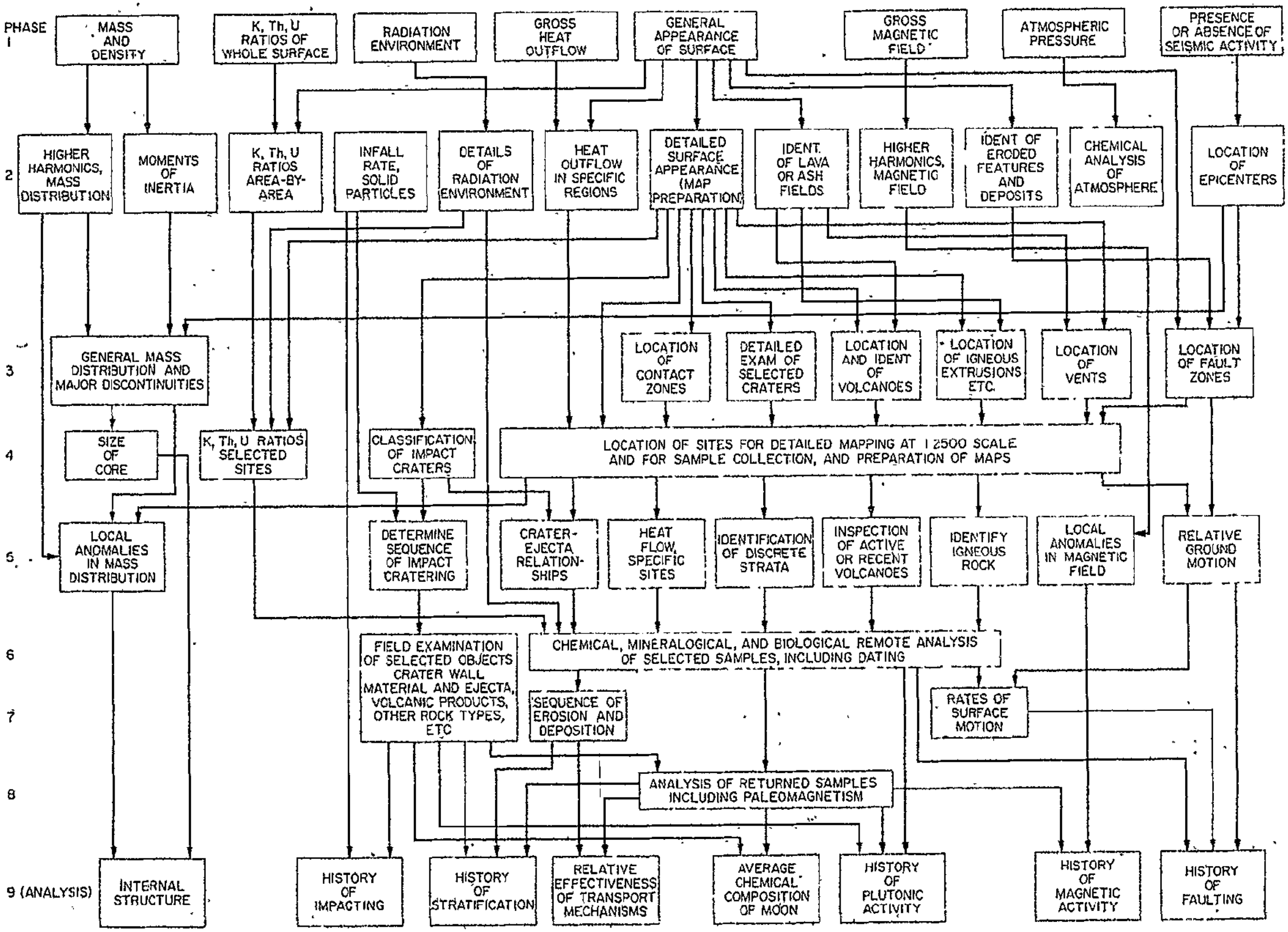

Fig. 1. Major lines of investigation

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'Cosmogony' refers to those lunar questions and measurements which bear on the Moon's history and constitution as a planetary body. These measurements are accorded high priority because they may yield information pertinent not only to the Moon itself, but also to the origin and evolution of the planetary system. In view of their special importance, these planetological measurements became the subject of a separate study within the JPL effort. The results of this study are contained in Ref. 4; their implications are discussed further below.

'Exobiology' refers to the class of experiments that can be done on the Moon in support of either techniques or theories in the search for extra-terrestrial life. It is ranked third on our list, not because its intrinsic interest is low, but because the Moon is expected to be a less rewarding target than other bodies for at least the early phases of the exobiology investigation.

'Astrogeology' is used here in a restricted sense, to refer to missions in the late, detailed and extensive phases of a program, similar to the fine-scale mapping and exploitation of unpopulated regions on Earth.

By using these categories as a guide, it is possible to place the various investigations in approximate priority order, and then to see whether or not a candidate flight program results in timely achievement of the most urgent measurements.

As mentioned above, special attention has been given to the planetological investigations. This was done, not only because both theory and observation indicate that the Moon itself may yield important planetological clues, but also because the results of lunar experiments may be critically important in planning investigations on other planets. In Ref. 4, five main questions were identified pertaining to the possible evolutionary paths of the solar system. These questions are as follows:

Q-1. Are the present individual planets and satellites chemically uniform or non-uniform?

Q-2. Did final accretion result in the present array of planets and satellites, or in a collection of bodies that were subsequently modified to yield the present array?

Q-3. Was the (circumsolar) dust cloud homogeneous at the time of final accretion?

Q-4. What was the state of the sun-cloud system when it first appeared as a recognizable entity?

Q-5. Were there large-scale elemental and isotopic nonuniformities in the (initial) contracted nebula?

The lunar investigations which bear most directly on these evolutionary questions are:

(1) Investigation of Moon's gross shape and mass distribution, including core if any.

(2) Investigation of heat flow, magnetism, and interaction with particle-and-field environment.

(3) Investigation of Moon's isotopic, chemical, and mineral composition.

The relationships between the major lines of investigation and the five evolutionary questions can be seen in Figs 2 through 6. Five shading patterns are given, each of which pertains to one of the five questions as listed at the left of Figure 1, and outlines those investigations expected to be most important for that question. Fundamental to all investigations is an adequate mapping program to confirm the validity and show the interrelations of the other measurements.

Once the priority measurements in the 'critical-data' and 'cosmogony' classes were identified, it was possible to begin the evaluation of the flight programs. This process is described in the following Section. The main conclusions of the scientific study were such as to require increased emphasis on (a) deep seismic probing, (b) selenodesy, (c) thermal, magnetic and particle-and-field experiments, and (d) measurements of radioactive-element ratios and extensive sampling and analysis of material collected on the Moon. By the same reasoning, relatively less importance was attached to local surface transport mechanisms, isolated tectonic effects, cratering mechanics, or other influences affecting lunar surface details. However, the study has continued to include consideration of these other effects as necessary elements in a complete program. Also special attention is being given to the subject of volatile constituents, especially water, because of the large effect that a discovery of available water would have on the later phases of the lunar exploration program.

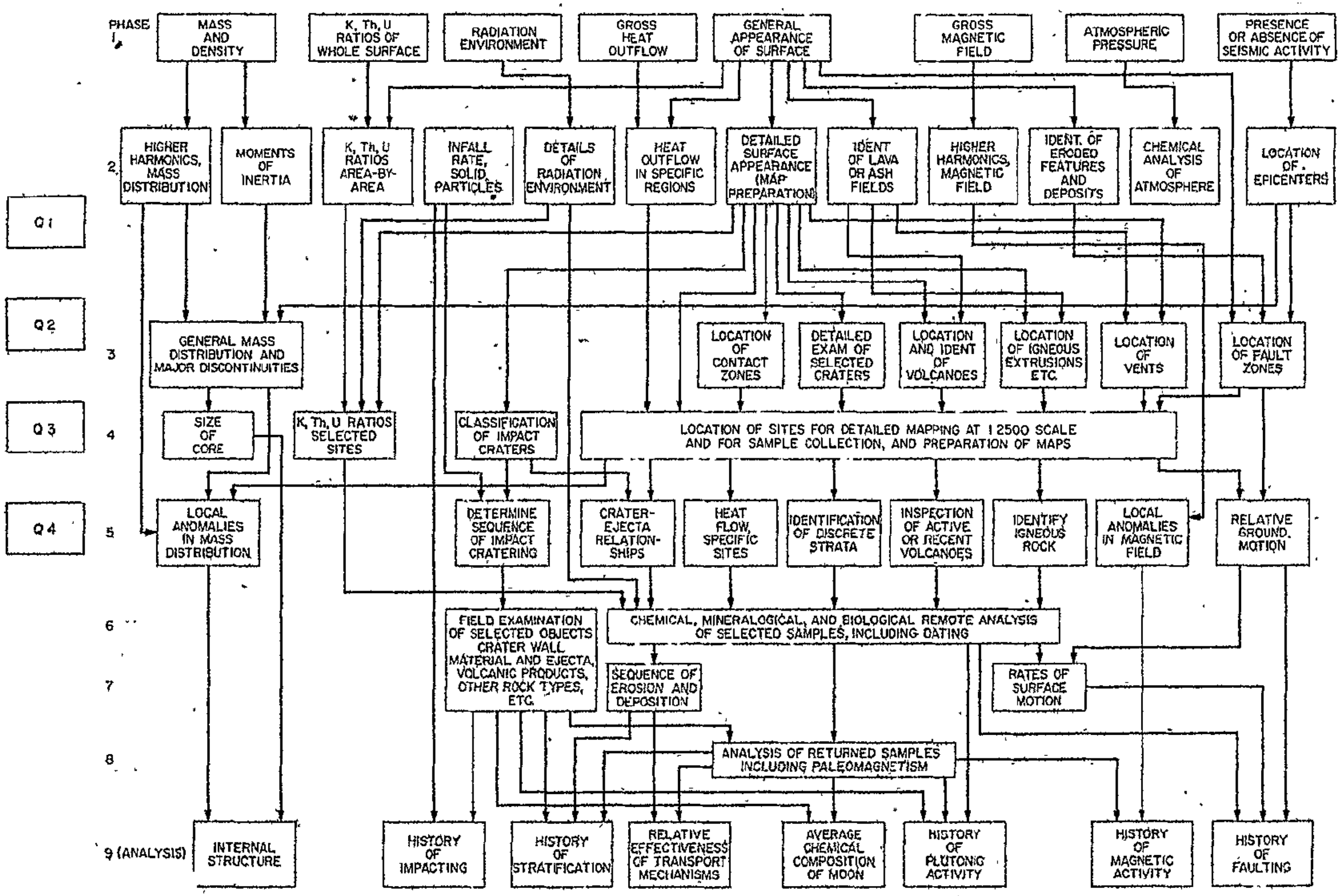

Fig. 2. Measurements pertinent to Planetology Question 1

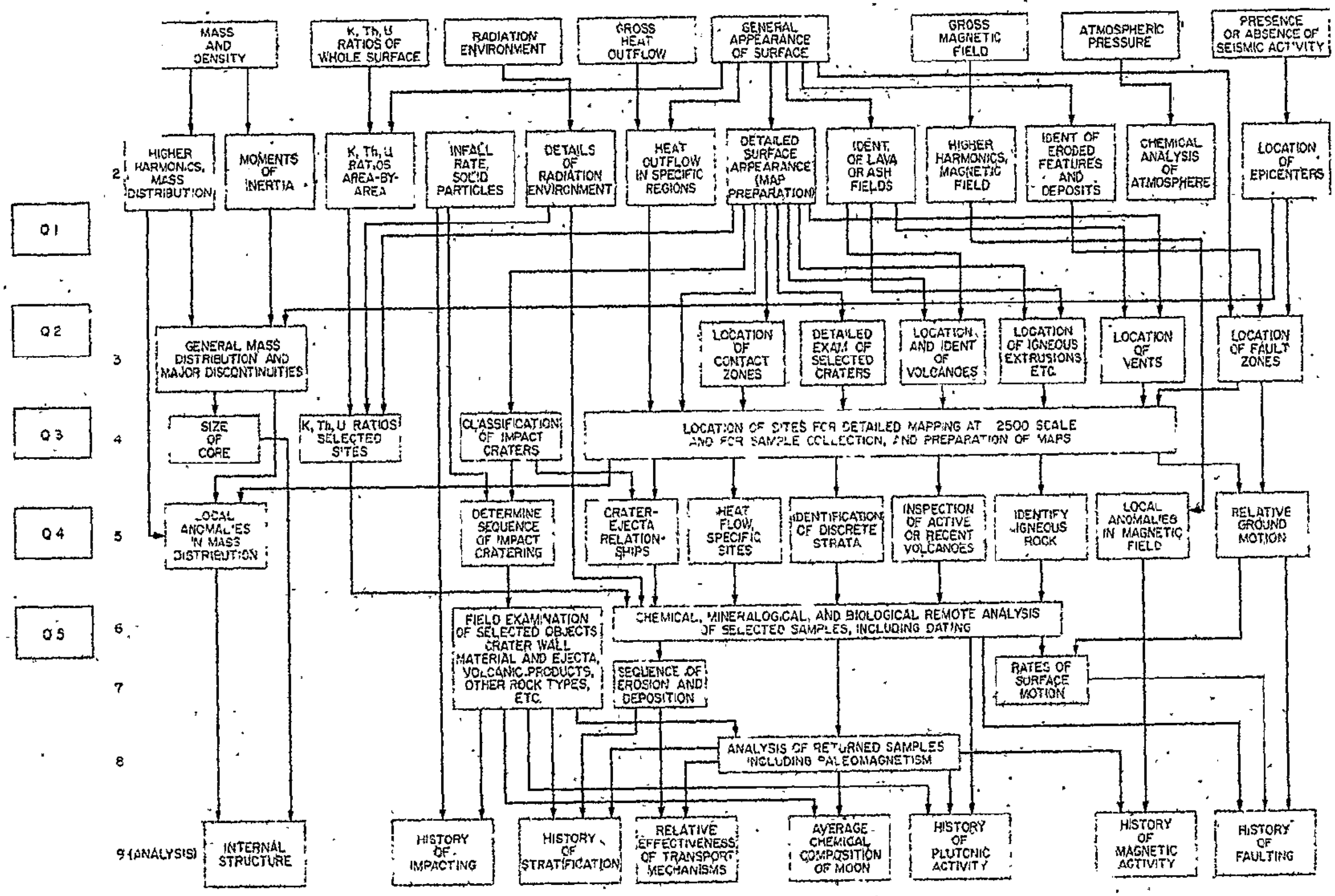

Fig. 3. Measurements pertinent to Planetology Question 2

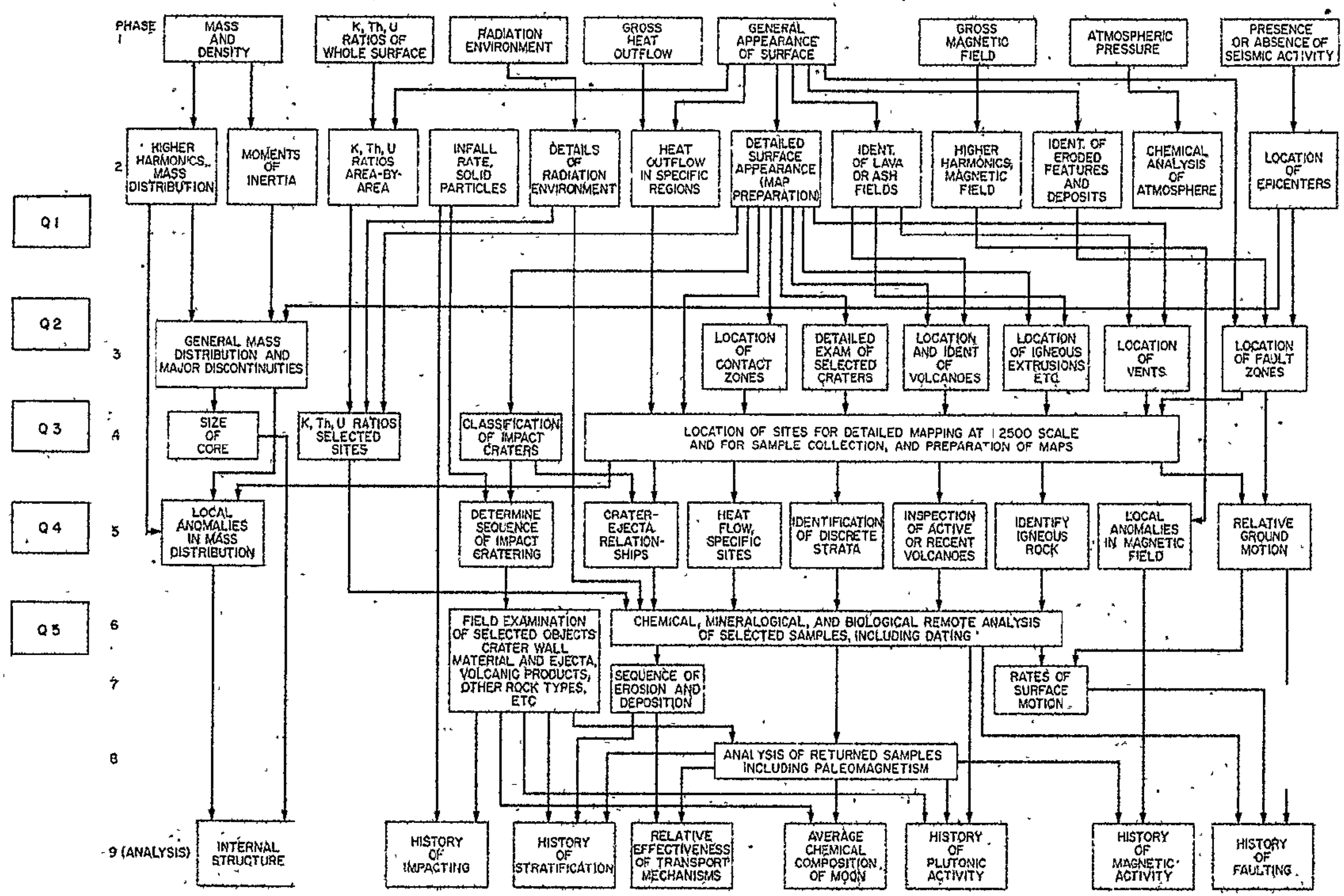

Fig. 4. Measurements pertinent to Planetology Question 3

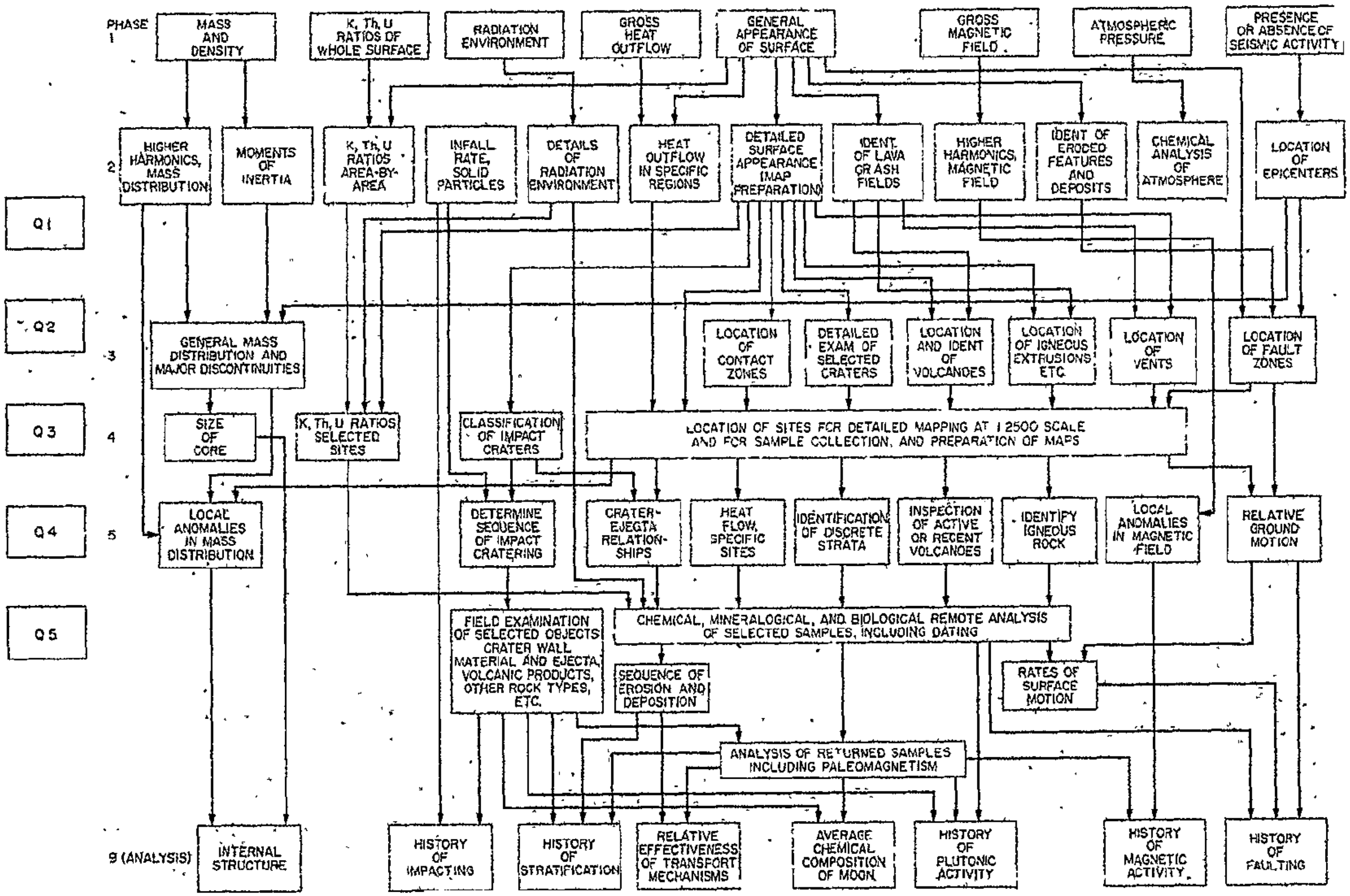

Fig. 5. Measurements pertinent to Planetology Question 4

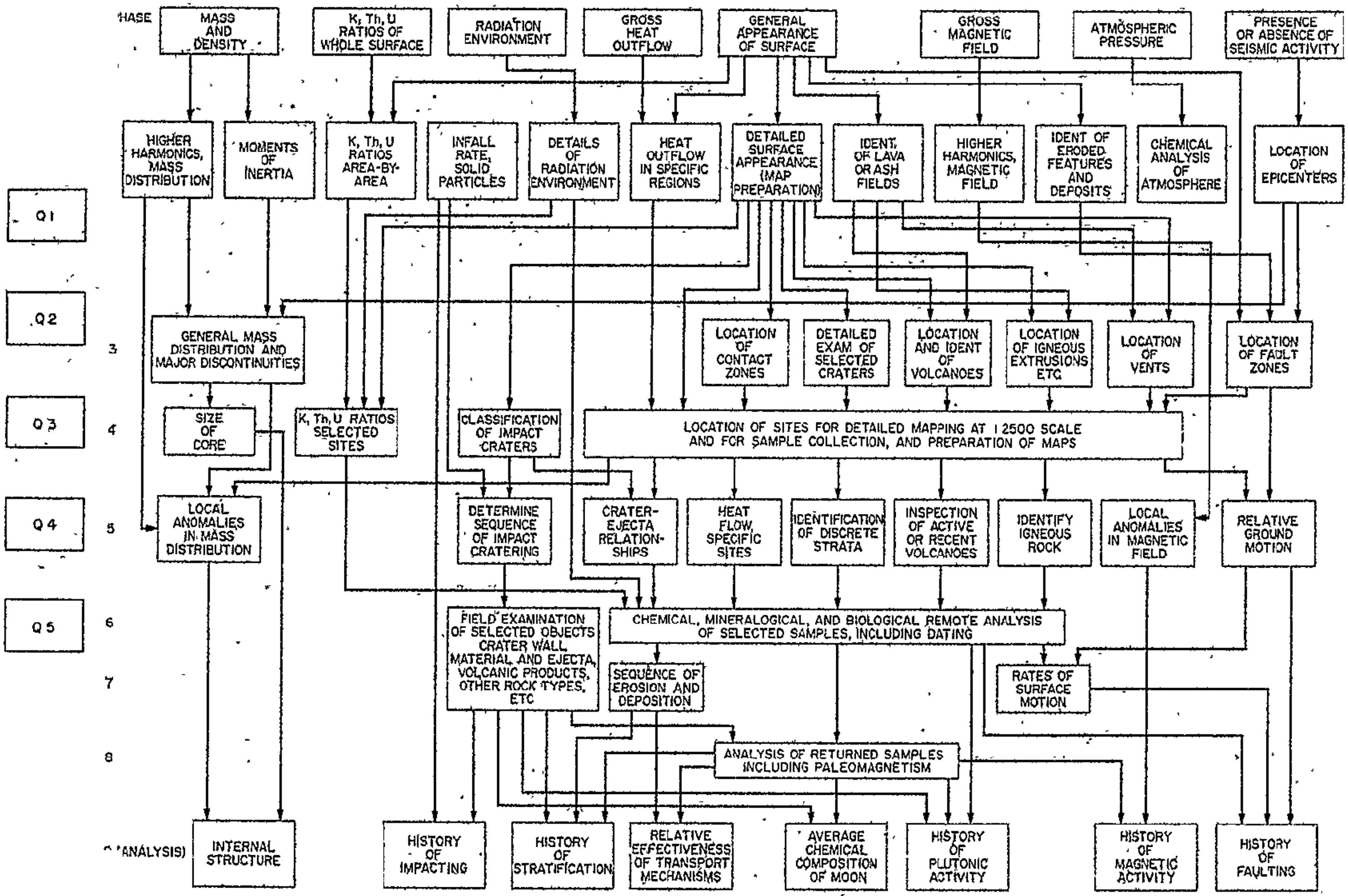

Fig. 6. Measurements pertinent to Planetology Question 5

III. MISSION PLANNING

Given the exploration rationa... ... ...rity goals described in the previous section, the next task in the study was to fit the required measurements into an orderly flight program. As a part of the previous work, instrument lists (Refs. 6 and 7) had been prepared, giving descriptions of the experimental equipment complements that could be carried by orbiting or landing spacecraft. With the aid of these lists and the measurement sets as listed in Ref. 1, it was possible to prepare a Mission Planning Chart, Table 1. In this Table the first column lists the Phases in the major lines of investigation as given in Fig. 1. The next column lists the measurement number, for cross-referencing with Ref. 1. The following columns list the objectives, instruments, and spacecraft believed applicable for each measurement, plus explanatory remarks, and the final column indicates the Space Science Board question or questions (see Table 2) to which each measurement is pertinent.

The major conclusion that can be drawn from Table 1 is that there are some high-priority lunar measurements which will not be made very soon if present flight-program plans are followed. If near-future lunar flights are limited to Apollo and modest extensions of Apollo, plus the remaining three Surveyors and one or two Lunar Orbiters, there will be no early opportunity to perform the following types of measurements:

(1) Simultaneous particle-and-field measurements on the surface and in orbit, including accurate mapping of the lunar magnetic environment.

(2) Operation of a long-baseline seismic net.

(3) Surface sampling, with on-site analysis and/or sample return to Earth, in Moon's polar regions and remote, rough areas.

(4) Subsurface probing in regions (e.g. polar areas) where theory or observation suggests possibility of finding recoverable water.

(5) Topographic mapping of selected areas in support of the above and other investigations.

The first class of measurements could be made by small orbiters (IMP or Pioneer-type spacecraft) operating in combination with ALSEP or Surveyor payloads on the lunar surface. The IMP or Pioneer spacecraft are preferable to larger spacecraft because of (a) their spin stabilization, which permits rapid measurement of field vector components, (b) their nonmagnetic design and construction, (c) their small mass and consequent small induced-radiation background, and (d) their relative independence of other mission constraints such as illumination, rendezvous

Table I. Mission Planning Chart (Sheet 1 of 8)

Phase	Measurement	Science objective	Science instruments (numerals indicate table numbers)		Spacecraft (■ Scheduled ☒ Planning) Unmanned: Orbiter I	Orbiter II	Surveyor	Landers	Rovers	Apollo: ALSEP	Geology	Sample return	AAP: LM/SS	IMP/Pioneer	ALSEP	LSSM	LFV	Base	Remarks	Major lines of investigation
I 1)	Three moments of inertia - range and doppler tracking of orbiter with transponder Orbit should be inclined to equator	BDA	I-5	Tracking of Orbiter	■	☒													Measurement is being carried out Radio or laser ranging from landers or ALSEP could provide additional data.	1(a)
			VIII	Tracking of IMP/Pioneer										☒						
2)	K, Th, U ratios of whole surface.	BDA	I-7	Gamma ray spectrometer	☒	☒													May be carried out on Orbiter 6, but if not, will not be done until early '70's with LM/SS or Orbiter Block II	1(c) 6(a) 5(a)
			VII-5	Gamma ray spectrometer									☒							
			VIII-2	Gamma ray spectrometer										☒						
3)	Gross radiation environment - high and low energy particle detectors such as electrostatic analyzers and gamma ray detectors in lunar orbit. Desire long lifetime and elliptical orbit.	AD	I	Radiation detectors		☒													These experiments are not being done. Expect to get data from a combination of landers and extrapolation of earth data.	3(e)
			VIII	Radiation detectors										☒						
4)	Gross heat outflow - best done in survey mission by polar orbiter.	BDA	I-9	Infrared radiometer		☒													Will not be done before early '70's on such vehicles as Orbiter Block II and LM/SS.	2(b) 6(a)
			I-11	Microwave radiometer		☒														
			VII-6	Thermal mapping									☒							
			VII-7	Microwave imager									☒							
5)	General surface appearance Orbiter and Lander photographs with orbiter resolution of 100 meters or better.	AD	I-12	Photo imaging	■														Measurement is being carried out	3(a) 4(a) 7(a) 3(b) 4(b) 7(b) 3(c) 4(c) 7(c) 3(f) 4(d)
			II-12	Survey camera			■													
6)	Gross magnetic fields - three-axis magnetometer on orbiter	AD	VIII-3	Vector magnetometer										☒					Can be done on unmanned photographic orbiter but IMP/Pioneer would do it best.	1(a) 2(d)
			I-1	Helium magnetometer		☒														
7)	Atmospheric pressure - red head in low altitude lunar orbit. Desire long lifetime.	A	I-13	Red head or trigger gauge		☒													Unless flown on Orbiter Block II, this measurement will not be accomplished; however, surface measurements during Apollo and AAP are scheduled.	3(d)

Table 1, Sheet 2 of 8

Phase	Measurement	Science objective	Science instruments (numerals indicate table numbers)		Spacecraft: ■ Scheduled ☒ Planning														Remarks	Major lines of investigation
					Unmanned					Apollo			AAP							
					Orbiter I	Orbiter II	Surveyor	Landers	Rovers	ALSEP	Geology	Sample return	LM/SS	IMP/Pioneer	ALSEP	LSSM	LFV	Base		
I 8)	Presence or absence of seismic activity - operation of instrument over period of at least six months and at several locations.	ABD	II-5	Single axis seismometer			☒												Will probably be done first as part of Apollo (ALSEP)	2(a) 3(f)
			II-17	Passive seismic				☒												
			V-1	Passive seismic*						■										
			IX-1	Passive seismic											☒					
II 9)	Higher harmonics of mass distribution.	BD	I-5	Tracking of Orbiter		☒													This measurement will probably be satisfied in the normal course of the lunar program.	1(a)
			II-22	Radio and laser ranging			☒	☒												
			VII-8	Gravity gradiometer									☒							
			IX-2	Laser-radio ranging											☒					
10)	Radial mass distribution - three-axis long-period seismometer net if moon has elastic wave source.	BD	II-17	Passive seismic				☒											Really need a net for this measurement which allows simultaneous data from widely separated geographic points.	1(b)
			V-1	Passive seismic						■										
			IX-1	Passive seismic											☒					
11)	K, Th, U ratios area by area - detail study of gamma ray distribution. Desire low-altitude polar orbit Measurement (2) may have satisfied this	DB	I-7	Gamma ray spectrometer		☒													It should be noted that if Orbiter 6 does not fly, this measurement may not be accomplished.	1(c) 5(a)
			VII-5	Gamma ray spectrometer									☒							
12)	Infall rates of solid particles - long-life micrometeorite detectors either in orbit or on surface.	DAB	I-4	Meteor dust detector	■	☒													Some information is being obtained by Orbiter Block I; however, this measurement requires data obtained from both orbiters and landers	3(b) 7(c)
			II-4	Micrometeorite ejecta			☒	☒												
			VIII-5	Micrometeorites										☒						
			IX-6	Micrometeoroid (ejecta)											☒					
			IX-13	Dust detector											☒					

Table 1, Sheet 3 of 8

Phase	Measurement	Science objective	Science instruments (numerals indicate table numbers)		Spacecraft: Unmanned — Orbiter I	Orbiter II	Surveyor	Landers	Rovers	Apollo — ALSEP	Geology	Sample return	AAP — LM/SS	IMP/Pioneer	ALSEP	LSSM	LFV	Base	Remarks	Major lines of investigation
II 13)	Details of radiation environment - high and low energy particle. Detectors on orbiters and ground stations. Ground stations would act as base for orbiter observations.	DAB	I	Radiation experiments		☒													These experiments are not being done. Will make some ground measurements as part of Apollo ALSEP.	3(e)
			II	Radiation experiments			☒													
			V	Radiation experiments						■										
			VIII	Radiation experiments										☒						
			IX	Radiation experiments											☒					
14)	Heat outflow in specific regions - this is a followup on measurement No. 4, includes both orbiter and ground measurements	DB	I-9	Thermal mapping		☒													If not accomplished in Phase I, this observation has high priority here.	2(b) 6(a)
			I-11	Microwave surface imager		☒														
			VII-6	Thermal mapping									☒							
			VII-7	Multifrequency microwave spectrometer									☒							
			V-6	Heat flow						■										
			IX-5	Heat flow in three-meter holes											☒					
			XII-9	Subsurface probe														☒		
15)	Detail surface appearance < 10 meters resolution and stereo of whole moon.	DA	I-12	Photo imaging		☒													Can be done by unmanned mapper as Orbiter Block II, and can be done by LM/SS Data to be achieved and retrieved only when needed.	2(a) 3(b) 4(a) 4(d) 7(c) 2(c) 3(c) 4(b) 7(a) 7(d) 3(a) 3(f) 4(c) 7(b)
			VII	LM/SS cameras									☒							

■ Scheduled ☒ Planning

Table 1, Sheet 4 of 8

Phase	Measurement	Science objective	Science instruments (numerals indicate table numbers)		Spacecraft (■ Scheduled ☒ Planning): Unmanned — Orbiter I	Unmanned — Orbiter II	Unmanned — Surveyor	Unmanned — Landers	Unmanned — Rovers	Apollo — ALSEP	Apollo — Geology	Apollo — Sample return	AAP — LM/SS	AAP — IMP/Pioneer	AAP — ALSEP	AAP — LSSM	AAP — LFV	AAP — Base	Remarks	Major lines of investigation
II 16)	Identification of lava and ash fields - surface composition measurements from lunar orbiter to be correlated with (15) above and ground truth data.	D	I-6	Radar		☒													The ground truth for this measurement can come from mineralogy-petrology measurements by landers, rovers, Apollo and AAP geology and sample return. It should be realized that these features may be recognized best by visual imaging.	4(c)
			I-7	Gamma ray spectrometer		☒														
			I-8	Thermal neutrons detector		☒														
			I-9	Thermal mapping		☒														
			I-10	X-ray fluorescence		☒														
			I-11	Microwave surface imager		☒														
			VII-4	Imaging radar									☒							
			VII-5	Gamma and x-ray spectrometer									☒							
			VII-6	Thermal mapping									☒							
			VII-7	Multifrequency microwave spectrometer									☒							
			VII	LM/SS imaging system									☒							
			I-12	Photo imaging experiment	■	☒														
17)	Higher harmonics of magnetic field - a three-axis magnetometer in polar orbit.	D	VIII-3	Magnetometer (vector)										☒					This measurement most probably will require an IMP/Pioneer spacecraft in combination with surface observations for base control. Best done by combination IMP/Pioneer plus ALSEP or combination IMP/Pioneer plus lander.	1(b) 2(d) 6(c)
			IX-4	Magnetometer											☒					
			II-9	Magnetometer									☒							
			V-2	Magnetometer						■										

Spacecraft: ■ Scheduled ☒ Planning

Phase	Measurement	Science objective	Science instruments (numerals indicate table numbers)		Unmanned: Orbiter I	Unmanned: Orbiter II	Unmanned: Surveyor	Unmanned: Landers	Unmanned: Rovers	Apollo: ALSEP	Apollo: Geology	Apollo: Sample return	AAP: LM/SS	AAP: IMP/Pioneer	AAP: ALSEP	AAP: LSSM	AAP: LFV	AAP: Base	Remarks	Major lines of investigation
II 18)	Identification of eroded surfaces and deposited layers - measurements from lunar orbiter to be correlated with (15) above and surface studies.	D	I-6	Radar		☒													Most of the data for this will probably come from visual imaging. Much information is being obtained by the Orbiter Block I series of spacecraft. Will need ground-truth for correlating orbiter data.	3(c) 4(b) 7(a)
			I-7	Gamma ray spectrometer		☒														
			I-8	Thermal neutrons detector		☒														
			I-9	Thermal mapping		☒														
			I-10	X-ray fluorescence		☒														
			I-11	Microwave surface imager		☒														
			I-12	Photo imaging experiment	■	☒														
			VII	Photo imaging experiment									☒							
			VII-4	Imaging radar									☒							
			VII-5	Gamma and x-ray spectrometer									☒							
			VII-6	Thermal mapping									☒							
			VII-7	Multifrequency microwave spectrometer									☒							
19)	Chemical analysis of the atmosphere - instruments for this should be carried on low altitude polar orbiter plus emplaced at possibly two ground sites.	DB	I-13	Quadrupole mass spectrometer		☒													The ground data for this measurement appears to be coming along; however, orbiter measurements will be difficult from the manned missions because of outgassing. This measurement is best done on a combination of Orbiter Block II and ALSEP. Also, measurement should be done early in lunar program.	2(c) 3(a) 3(d) 8(a)
			II-14	Gas chromatograph				☒												
			II-18	Lunar atmosphere experiment				☒												
			V-4	Ion detector						■										
			V-5	Pressure gauge						■										
			VI-4	Sample return								■								
			IX-8	Ion mass spectrometer											☒					
			IX-11	Lunar atmosphere experiments											☒					

Phase	Measurement	Science objective	Science instruments (numerals indicate table numbers)		Unmanned: Orbiter I	Unmanned: Orbiter II	Unmanned: Surveyor	Unmanned: Landers	Unmanned: Rovers	Apollo: ALSEP	Apollo: Geology	Apollo: Sample return	AAP: LM/SS	AAP: IMP/Pioneer	AAP: ALSEP	AAP: LSSM	AAP: LFV	AAP: Base	Remarks	Major lines of investigation
II 20)	Location of epicenters - three long-period, three axis seismometers separated by hundreds of kilometers and operating simultaneously.	BD	II-17	Passive seismic				☒											The three scheduled Apollo ALSEP missions will give a start on this; however, a reasonable solution to this measurement will require many years of seismic-net operation. Will probably also require a tripartite net on backside.	3(j) 7(b)
			V-1	Passive seismic						■										
			IX-1	Passive seismic arrays											☒					
III 21)	Physical libration -	DB	II-22	Ranging (radio and laser)				☒												1(a) 1(b) 6(b)
			IX-2	Laser-radio ranging network											☒					
22)	Radial discontinuities in mass - if moon is seismically active, seismic measurements in Phase II may show the core-mantle discontinuity. If moon is not seismically active this measurement will require large-scale active seismic experiment and must be delayed to a later phase.	BD	II-17	Passive seismic				☒											If moon is seismically active and considering the long lifetime for both the Apollo and AAP ALSEP, we should have good information on this measurement by the middle '70's	1(a) 1(b) 1(c) 2(a)
			V-1	Passive seismic						■										
			IX-1	Passive seismic arrays											☒					
23)	Location of contact zones - visual examination from the surface of specific regions selected on the basis of Phase II maps. Roving vehicle traverses probably required - field geology.	D	III	Roving vehicle science					☒										The data for selecting these sites is probably available from ground based telescopic observations and Orbiter I data; however, Orbiter Block II or LM/SS images will certainly enhance site selection. The Apollo geologist may observe this type of feature; however, specific mission to observe contact zones will probably not be accomplished until AAP.	4(b) 4(d) 7(a) 7(b) 7(d) 3(a) 3(b) 3(c)
			VI	Apollo-geology sample return						■	■									
			X	LSSM science												☒				
			XI	LFV science													☒			
			XII	Lunar base studies														☒		

■ Scheduled ☒ Planning

Table 1, Sheet 7 of 8

Spacecraft: ■ Scheduled ☒ Planning

Phase	Measurement	Science objective	Science instruments (numerals indicate table numbers)	Unmanned: Orbiter I	Unmanned: Orbiter II	Unmanned: Surveyor	Unmanned: Landers	Unmanned: Rovers	Apollo: ALSEP	Apollo: Geology	Apollo: Sample return	AAP: LM/SS	AAP: IMP/Pioneer	AAP: ALSEP	AAP: LSSM	AAP: LFV	AAP: Base	Remarks	Major lines of investigation
III 24)	Detailed examination of selected craters - ground study of classic crater types. For the larger craters and any detail study, this observation will require considerable mobility and long stay times - field geology.	D	III Roving vehicle science					☒										This measurement is only marginally within the capabilities of the AAP stay times and the range of the LSSM and LFV. The Apollo astronaut will accomplish some preliminary observations on small craters in the landing area	3(b) 3(c) 4(b) 4(d) 7(a) 7(c)
			VI Apollo-geology and sample return							■									
			X LSSM science												☒				
			XI LFV science													☒			
			XII Lunar base studies														☒		
25)	Location and identification of volcanoes - includes ground studies of select sites chosen from Phase II maps - field geology	D	III Roving vehicle science					☒										Orbiter mapping data for study of these sites can be provided from the Orbiter Block II or LM/SS spacecraft but specific sites may not be visited before LSSMs and LFVs are available. Will likely involve traverse over rough ground.	3(a) 4(c) 4(d) 4(a)
			X LSSM science												☒				
			XI LFV science													☒			
			XII Lunar base studies														☒		
26)	Location of igneous extrusions - includes ground studies of select sites chosen from Phase II maps, field geology	D	III Roving vehicle science					☒										Same as above.	3(a) 4(c) 4(a)
			X LSSM science												☒				
			XI LFV science													☒			
			XII Lunar base studies														☒		
27)	Location of gas vents - includes ground study of some sites - field geology.	D	III Roving vehicle science					☒										Location of some sites from orbiter visual data and remote sensors. In addition to the geology of classic areas would also like to study any gas emissions. This latter task will probably require the use of LSSM or more probably LFV to emplace instruments.	3(a) 3(d) 4(h)
			X LSSM science												☒				
			XI LFV science													☒			
			VII Lunar base studies														☒		
			IX-8 Ion mass spectrometer											☒					
			IX-11 Lunar atmosphere experiment											☒					

Table 1, Sheet 8 of 8

Phase	Measurement	Science objective	Science instruments (numerals indicate table numbers)	Spacecraft (■ Scheduled ☒ Planning) Unmanned					Apollo			AAP							Remarks	Major lines of investigation
				Orbiter I	Orbiter II	Surveyor	Landers	Rovers	ALSEP	Geology	Sample return	LM/SS	IMP/Pioneer	ALSEP	LSSM	LFV	Base			
III 28)	Location of fault zones - includes ground studies of selected features, field geology	D	III Roving vehicle science					☒											Location of some of these features will be accomplished from Phase II mapping, but others will require detail ground mapping by a geologist to locate them. Ground studies will probably require considerable mobility and long stay times; also, active seismic or other geophysical techniques will be useful.	3(f) 4(a)
			X LSSM science												☒					
			XI LFV science													☒				
			VII Lunar base studies														☒			

Table 2. Relation between major lines of investigation and the fifteen SSB questions

1. Body Structure
   (a) Mass Distribution (1, 2)
   (b) Presence of a Core (1, 3, 12)
   (c) Gross Differentiation (1, 3, 10)
2. Body Activity
   (a) Seismic (1, 3, 5)
   (b) Thermal (1, 3)
   (c) Volcanic (3, 6)
   (d) Magnetic Field (1, 3)
3. Surface Activity
   (a) Volcanism (3, 5, 6, 8)
   (b) Impacts (5, 7, 13)
   (c) Erosion, Transport and Deposition (7)
   (d) Escape of Gases (3, 8)
   (e) High Energy Radiation (14)
   (f) Seismic Activity (5, 6, 11)
4. Surface Morphology
   (a) Appearance (front and back) (2, 5, 6, 7, 12, 13)
   (b) Identification of Strata (1, 5, 6, 7, 10, 12, 13, 14)
   (c) Lava, Volcanoes and Volcanic Rocks (3, 4, 5, 6, 8, 12)
   (d) Crater and Rock Distribution (6, 7, 10, 13)
5. Surface Composition
   (a) Chemical (4, 6, 8, 9, 10, 12, 13, 14)
   (b) Mineral (4, 6, 12, 13, 14, 15)
6. Body History
   (a) Thermal (3, 5, 6, 12)
   (b) Interaction with Earth (11)
   (c) Magnetic (3, 15)
7. Surface History
   (a) Stratigraphy (1, 5, 6, 7, 10, 12, 13, 14)
   (b) Seismic Activity (5, 6, 11, 12)
   (c) Impact Rates (6, 10, 13)
   (d) Igneous Rock (6, 10, 12)
   (e) Magnetic Remnants (10, 15)
8. Organic Activity
   (a) Current (9)
   (b) Remnants (9)

maneuver criteria, and so forth. Because of these advantages it has been recommended that IMP or Pioneer-type spacecraft be carried aboard Apollo or AAP vehicles and ejected after reaching lunar orbit. Also a study of the experimental criteria for such missions has been made; the results are given in Ref. 5.

The second class of measurements requires (a) landings at three or more widely-separated points on the lunar surface, and (b) simultaneous operation of the landed seismometers over a time period reasonably long compared with the (now unknown) frequency of natural seismic events. These requirements suggest use of Surveyors in combination with ALSEP. Use of ALSEP alone would require manned flights scheduled closely so that the last seismometer would be delivered while the first one still had several months of life remaining.

The third class of measurements, surface sampling, presents special problems. Apollo and AAP flights are expected to yield adequate sampling data within their areas of access. For Apollo these areas will be equatorial, flat mare sites selected primarily for safety, with sample collection limited to a region within about one kilometer of the landing point. For AAP the constraints are less definite at present. There is a possibility of reaching the middle latitudes of the Moon, and also a possibility of landing close enough to steep, rough or unique features (e.g. Copernicus) so that, with the aid of transport vehicles, astronauts can reach the sites of interest. The two kinds of transport vehicles being considered for AAP are the Local Scientific Survey Module (LSSM) and the Lunar Flying Vehicle (LFV), whose expected radii of action are a few km and a few tens of km, respectively.

It now appears likely that Apollo and early AAP sampling data may not be definitive for the whole Moon, because of the limited regions and durations of access. Therefore the concepts of longer-range surface exploration vehicles have been examined. These vehicles fall into two classes: (1) manned, traveling laboratories (MOBEX, MOLAB) and (2) unmanned, instrumented, sample-collecting mobile machines. Vehicles in the first class would require so much advancement of technique that many years would pass before they could be in use upon the Moon. Vehicles in the second class would also require significant new engineering development but could be available sooner and at much smaller cost. For example it may be possible to develop an automated version of the LSSM. (An automated LFV is conceivable but appears impractical because of navigation and guidance problems). Possibly even more useful, for reasons to be outlined in the next Section, would be a machine

somewhat smaller than the present LSSM design. Such a machine would fulfill the role of a "field assistant" as discussed in various conferences on the geological exploration of the Moon.

The fourth class of investigations listed above (subsurface probing and search for aquifers) may require a major engineering development in deep drilling on the Moon. Alternatively it may merely require that a Surveyor or Apollo-class spacecraft, carrying one or another of the drills already being tested, be landed in a high-latitude mare-border region. (See, for example, photographs from Lunar Orbiter 4, such as Frame 157-H, right.) If conditions are favorable ice may be trapped within a few centimeters of the surface near the poles. However, a slight reduction in trapping efficiency (or a major shift in the orientation of the Moon's poles during geologic time) would be enough to place the ice boundary deep below the surface. Therefore, preliminary experiments, such as neutron spectrum measurements from orbit (see Ref. 5), polar exploration by "field assistant" rovers, or the previously-mentioned polar landings of instrumented Surveyors, should be considered.

The fifth class of activities (metric mapping), though not firmly specified in the approved program plan, is recognized as a necessary adjunct to surface exploration. It may be partly achieved through AAP lunar-orbital flights carrying cameras such as the Lunar Mapping and Survey System (LMSS). However, an extension of the present Lunar Orbiter series, with some engineering and calibration improvements, also appears necessary for the economical achievement of the required mapping coverage. Development of a new unmanned photographic orbiter is feasible but not essential; the present spacecraft design (adapted, if necessary, to Centaur launch if Agena is phased out) could be modified and used.

IV. PROGRAM IMPLICATIONS OF THE STUDY

The JPL study to date has not identified any items in the current, approved NASA flight program plan that are unnecessary or scientifically undesirable for the lunar exploration rationale represented by Fig. 1. It has--as described in the previous Section--identified some additional lunar missions, not included in the present approved program, which would be highly desirable on scientific grounds. Some of these missions (e. g., Surveyor seismic and IMP/Pioneer particle-and-field experiments) would be only minor extensions to the present program. Others (e. g., orbital metric mapping and Surveyor polar landings) would involve substantial spacecraft procurements but could and should be done as straightforward applications of existing technique. Still others (e. g., remote-area roving sampling, and deep drilling if required) would require new engineering developments. In order to gain an understanding of the problems presented by the remotely-controlled rover or "field assistant" mission, the JPL group has started an engineering study of surface vehicles. In order to put reasonable bounds on the rover configurations to be studied, the following logic has been applied:

(1) The most urgent remote-area sampling investigations can probably be done by an instrument payload (see Ref. 7) similar to the payloads originally intended for Surveyor; namely, a coordinated group of collection and sensing devices with an aggregate weight of 100 to 200 pounds. Some of this weight would be devoted to imaging and on-site chemical, mineral, and isotopic sensing, but some of it would also be used for storing collected samples which the rover would eventually bring to a rendezvous site for retrieval and return to Earth. In the present study the return is assumed to be carried out by a manned vehicle; unmanned sample-return systems, as previously studied for Surveyor, could also be considered if necessary.

(2) Previous studies indicated that a separable rover, delivered as payload by Surveyor and hence limited to a <u>total</u> mass of 100 to 200 pounds, could be useful in local surveys, and prototype vehicles (called SLRV) were built. However, it was recognized that such small vehicles, with a <u>payload</u> mass of only 10-30 lb., would have marginal performance for the long-range sampling mission.

(3) A reasonable upper limit on size for an automated rover would be that of the LSSM, whose mass is about a thousand pounds and whose size is such that it must be delivered to the lunar surface by a Saturn-V-launched, unmanned Lunar Module, as a part of the AAP dual-launch mission concept.

(4) Between these lower and upper size limits, there is probably a feasible vehicle design which can perform the most urgent sampling measurements, while still being small and light enough to be delivered (a) as an integral package ("Surveyor-on-wheels") launched by Centaur, (b) as payload aboard a single-launch, manned AAP mission, or (c) as a separable payload item aboard an unmanned soft-lander launched by a vehicle intermediate in size between Centaur and Saturn V. (A fourth delivery mode--launch from an orbiting AAP-class manned spacecraft--might be of interest but has not yet been considered in detail.) Each of these three delivery modes is compatible with a roving vehicle in the 600-800 pound mass range, with the requisite instrument payload of 100 pounds or more; i.e., with an investigative capability similar to that originally planned for Surveyor, but augmented by (a) the ability to deliver material samples for return to Earth, and (b) the ability to take the samples from particular selected spots along a traverse, rather than just from a single, random point.

Though all of the three delivery modes mentioned above would give about the same gross mass (600-800 lb.) for the roving vehicle itself, there would be significant differences in performance, cost, and development time. The first mode (integral vehicle design, Centaur launch) is the least costly and represents the smallest departure from existing technique. But the performance of the rover is constrained by (a) Centaur injection mass capability and (b) incorporation of the guidance, control and propulsion subsystems required for transit to and landing on the lunar surface. The parts of these systems which can not be used after landing for navigation or locomotion become surplus and have either to be jettisoned or carried as excess burden during the roving mission.

The second delivery mode (rover as payload on manned LM) eliminates the need for transit and descent subsystems on the rover, enhancing the payload potential for any given gross rover mass. But it does couple the rover to the manned flight schedule, it requires the solution of new interface problems on the

LM, and it competes with other LM payload items such as a drill or other instruments, or expendables for lengthening the astronauts' stay on the Moon. The cost of this mode is likely to be higher than that of the first mode, unless the LM-associated costs are excluded.

The third delivery mode (rover as payload on an unmanned lander about three times the size of a Surveyor) offers the greatest flexibility in design of the "field-assistant" mission but it requires not only the new engineering development of the rover itself, but also the development of a new lunar-soft-landing spacecraft and the adoption of an intermediate-size launch vehicle (such as a Saturn I or Titan III with upper stage) not previously used in the lunar program, and so is likely to be the highest-cost option.

In addition to the cost differences, there is another effect that must be considered in comparing modes. In each lunar project to date, as development difficulties were encountered, scientific objectives were compromised. This experience suggests that any future projects having primarily scientific goals should incorporate only a minimum of new engineering development. For the "field assistant" mission, there is no escaping the need for locomotion and navigation on the Moon. These are new engineering requirements, and the development program should be laid out so that they have a chance of being met with a minimum compromise of mission goals, and so that after the technical problems are mastered, enough resources remain to complete the scientific program. Of the modes considered, the Centaur-launch mode appears least demanding of new development and hence has been selected for the baseline rover design to be analyzed in the JPL study. The automated LSSM might also be a candidate, but it does require the development of the unmanned LM and the dual-launch AAP mission profile. If a satisfactory sampling mission can be executed by the smaller rover with only one Saturn V launch (to return the collected samples to Earth) it is clearly the least costly and the least demanding of new developments, of all the options examined.

To determine whether or not a rover near the baseline size is practical for the mission considered, some design studies have been made. Results of these studies are given in the following Section, and also have been summarized briefly in Ref. 8. It does appear that a useful mission can be achieved by a spacecraft compatible with present Centaur performance; if planned Centaur improvements are effected, significant enlargements of rover payload can be made. The next step in

the study is intended to include a variation of parameters about those of the baseline design, and a further investigation of the salient technical problems of small vehicles roving on the Moon.

V. MOBILE VEHICLE DESIGN STUDY

A. Mission Objectives

The primary objective of the roving mission is the collection of lunar material samples for return to Earth. The samples must be identified as to location and as to the geologic situations in which they are found. A minimum goal consists of (a) collection of 20 to 100 samples of naturally particulate lunar matter, each having a volume of 1 cc, over an extended traverse, (b) imaging (with Surveyor TV resolution and photometric fidelity) of the surroundings at each sample site, and (c) transport of the collected samples to a point suitable for retrieval. In present studies this would be a manned landing site; unmanned sample return is another possibility,

To the above minimum goal, if circumstances permit, the following additional goals should be added:

(1) On-site chemical and mineral analysis.

(2) Collection from sources (e.g. rocks) that are not already particulate, and hence need to be chipped or drilled.

(3) Collection from beneath the surface.

(4) Detection of volatiles, e.g. water vapor or ice.

In order to perform the above mission the rover must have telecommunicatio and navigation subsystems. These can be used for additional benefit to the scientific mission. For example, incorporation of an S-band ranging transponder, as used in Lunar Orbiters, will permit precise location of points on the lunar surface and—during times when the rover is not traveling—measurements of lunar motions.

By using the navigation subsystem in combination with the imaging subsystem, local surveys can be made during the traverse missions, for such purposes as selection of additional areas for manned landings and manned surface expeditions. When taken together with the primary goals listed above, these requirements suggest that the imaging system should provide stereo views (either via multiple sensors or by vehicle motion) from as high a vantage point as practical and that the navigation subsystem should permit absolute location of the rover to within a few km, and relative location (i. e., with reference to previous overhead photo coverage) to within tens of meters with respect to known prominent objects in the rover field of view.

B. Lunar Model for Rover Design

In order to make progress with a baseline design some limits must be set on the surfaces that the vehicle is expected to traverse reliably. It is recognized that much more analysis can be and will be done, now that Surveyor and Lunar Orbiter data are in hand; therefore for the initial stage of the present study only a simple and rather general lunar model has been assumed.

The main property easily seen in the data is that finely divided granular material is dominant on the surface of the Moon. There is no definite indication of large bedrock outcrops in any of the photos now available. Fissures and cavities appear to have been filled by the granular material. Though some of the boulders observed on the surface may be merely impact-compacted, friable clods, others are surely large and hard enough to be real obstacles. For example two instances were found of large (house sized) boulders strong enough to survive rolling down the inner wall of a crater. Clearly there is no reasonable small rover design that could traverse a strewn field of such boulders without detours. Therefore climbing over large, steep objects, across crevasses, and so forth, is not a dominant requirement; the rover must instead be guided around such obstacles.

The machine should routinely and safely travel up, down, or across reasonably steep slopes; not only because such regions (e.g. central peaks in craters) are expected to be scientifically interesting, but also because moderate slopes will be encountered at random along any traverse.

There appears to be no point in attempting to traverse the Moon's steepest slopes, some hundreds of which can be seen from Earth to be greater than the angle of repose for dry, granular, non-interlocking materials (about 34 deg). The reason is that no vehicle design could give positive protection against avalanches on such slopes. Pending more accurate analysis and tests, the baseline vehicle has been designed for static stability on slopes up to 40 deg, giving a probably realistic slope-crossing limit on soft soils in the region of 20-25 deg. This is sufficient for most of a normal traverse without special command and control. In steeper areas progress would be made only after deliberate decisions by the operator. In order to avoid detouring around small obstacles, the machine should have good bump or dip and straddle clearances; the baseline design has a nominal clearance greater than 30 cm.

Since large, steep objects, which the vehicle can neither climb nor straddle, are expected to be of scientific interest, the sample-collection equipment must reach

not only beneath the vehicle but also out in front of it. A Surveyor-type sampler could be used, or two separate special-purpose devices (e.g. a central drill and an end-mounted percussion sampler) could be devised. For either case the appropriate lunar model is assumed to be simply a vertical wall of indefinite extent, from which samples are desired up to the greatest practical height.

Surveyor data indicate that soil bearing strength should pose no serious mobility problems; surface temperature extremes do present difficulties in wheel design, and for present purposes the design study is being based on the maximum temperature excursion observed in the equatorial regions of the Moon.

C. Design Criteria for Mobile Vehicles

In order to establish the preliminary mission objectives and design requirements for an unmanned lunar roving vehicle it is necessary to consider appropriate science and technology of the roving vehicle to be used in combination with the various lunar delivery systems available for lunar exploration.

The cislunar transport and soft landing delivery modes being considered are:

(1) Unmanned spacecraft carrying rover as payload.

(2) Integrated unmanned spacecraft incorporating transit, descent and mobility subsystems.

(3) Manned Lunar Module (LM) carrying rover as part of payload.

(4) Unmanned version of LM carrying larger rover (e.g., automated LSSM).

(5) Unmanned rover launched to lunar surface from orbiting spacecraft.

Recognizing the importance of flexibility to the lunar program, it was decided to explore the possibility of an adaptive roving concept, variations of which could be used in combination with any of these modes of delivery.

In order to take advantage of previous roving vehicle developments a review was made of the Surveyor Lunar Roving Vehicle (SLRV), a six wheeled minimal type of roving vehicle (See Fig. 7) which was to be attached to the side of the Surveyor as its full payload (about 100 lb). Upon release this vehicle was capable of roving a few km in the landing area to provide limited local information. Its purpose was to extend the lunar science and Apollo site certification capability of the stationary Surveyor. This rover was first studied in 1964 as a part of the Surveyor follow-on program and its development was carried through the working model evaluation phase. The SLRV prototypes were designed to be highly mobile over unknown terrain.

Fig. 7. Surveyor Lunar Roving Vehicle (SLRV) during test

They were, however, very limited in science capability (about 10-15 lb. of instruments including imaging). Study efforts were then made to improve this condition by eliminating some of the duplication of subsystems which occurred within the Surveyor and SLRV combination. This improved integration (mainly of communications systems) increased the roving weight to about 200 lb., giving a somewhat increased capability of the roving vehicle. However, the effect on payload was not accurately evaluated before the study was halted.

In considering the results of these studies it was concluded that even the improved SLRV approach was marginal from the science standpoint and that consideration should be given to concepts having increased capability. Previous studies had shown that a desirable payload range was approximately that of the Surveyor (50 to 100 lb.), and ways to use up to 200 lb. could be foreseen.

To achieve this capability it was decided to explore a totally integrated design concept adapting the appropriate Surveyor transport and descent systems to soft land a roving vehicle on its wheels. At the same time it was considered desirable to provide for the detachment of these transport and descent systems for adaptability to the various lunar delivery modes mentioned above.

It was decided to first explore a reference or base line design based on Surveyor technology. This size was chosen since accurate and detailed performance information was available on the flight tested transport and descent systems of the Surveyor. Also, extrapolation from the hundred pound SLRV indicated that a 450 to 500 lb. integrated rover as indicated in Fig. 8 could probably provide an adequate payload capability for this roving mission. This figure presents the weight of the

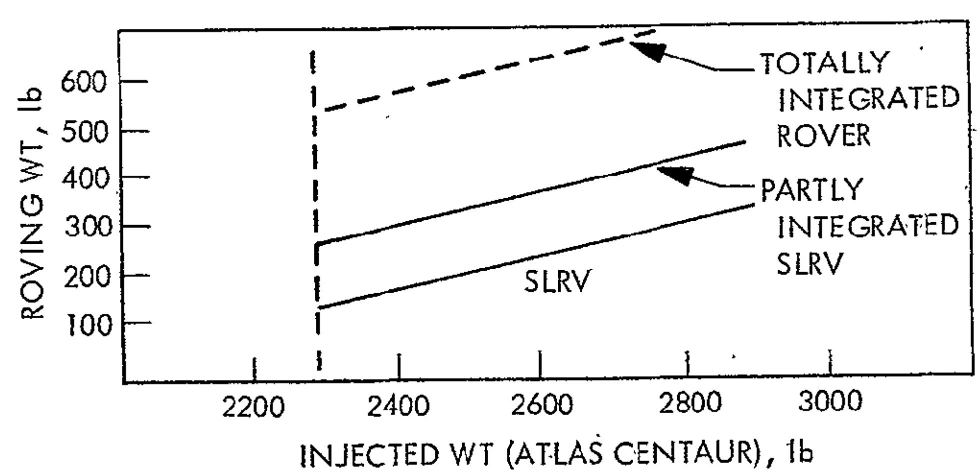

Fig. 8. Rover weight extrapolation

various designs of integrated roving vehicle as a function of the injection capability of the Atlas-Centaur. Of course, the totally integrated concept as well as the partially integrated SLRV's could benefit from any of the performance improvements now being considered for the Atlas-Centaur, some of which would result in injected weights well above 3000 pounds.

Configurations considered included walking as well as 3, 4, and 6 wheeled designs. The 4 wheel concept chosen was considered very desirable from the roving performance and descent systems integration standpoint, to serve as a baseline design. More detailed investigation could change this choice.

The following discusses the design criteria developed for the delivery systems considered. It should be noted that the delivery by the integrated (mode 2 above) system is presented in some detail since this approach requires the most extensive descent system for landing the roving vehicle. Delivery of this same lunar roving vehicle by the separate automated lander or by manned spacecraft essentially involves the following variations of the descent system plus adaption to the booster, LM, or CSM constraints.

(1) Manned LM or automated lander-The same lunar rover with the descent systems detached.

(2) Descent from CSM in Orbit-The same lunar rover and Surveyor type descent system incorporating a pitch turn programmer and substituting a smaller main retro motor.

D. Baseline configuration

1. Integrated spacecraft design. The design configuration developed for this vehicle adapts the major transport and descent systems of the Surveyor to soft land a four wheeled roving vehicle on the lunar surface. See Fig. 9. It uses the Surveyor propulsion systems (vernier system and main retro motor) in combination with Surveyor radar systems (AMR and RADVS) for the descent requirements. However, it repackages the other Surveyor equipment into a single compartment above the main retro motor rather than distributed around it as in the Surveyor. This design provides an efficient body and chassis for the roving vehicle with a low center of gravity for good landing and roving stability.

This central equipment compartment also provides a thermally controlled environment for the experiments and electronics and supports four wheels with

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independent springing for landing and roving. For launch these wheels are rotated to a stowed position around the main retro motor to fit within the Centaur shroud. A separable "A" frame, located on top of the central compartment, carries the major descent systems including:

(1)	Vernier Propulsion System	- Including appropriate electronics - (auto pilot, etc.)
(2)	Radar Descent System	- Antennae plus appropriate electronics
(3)	Canopus Sensor	

The articulated solar panel and antennae (high gain and omni) are located above the central compartment. These are arranged to permit the detachment of the "A" frame and its descent systems equipment from the rover for the LM or CSM delivery modes. This also permits the jettisoning of this expended equipment after landing for the unmanned delivery mode. The main retro motor, which is jettisoned during descent in the normal Surveyor fashion, is located below the compartment. It contains the altitude marking radar (AMR) in the nozzle in the normal Surveyor manner along with the flight controls (electronics and gas attitude control system) which become expended equipment at the start of descent. Both are ejected upon start o: the main retro motor as in the standard Surveyor AMR ejection.

Stereo imaging sensors plus a ranging sensor are located above the compartment for prime data and roving navigation while the other experiments are located within the compartment with access to the side or bottom as required.

A preliminary estimated weight breakdown of this design concept is shown o: Table 3. This estimate indicates that the rover payload is generally comparable t that of the Surveyor. However, it should be emphasized that these are only initial exploratory estimates and that accurate performance will be determined from the more complete study now in progress.

To provide a gross performance perspective of this unmanned class of roving vehicles, rough performance estimates were also made of larger roving vehicles. This result is shown on Fig. 10 which presents roving payload as a function of injected or roving weight capability of the launch vehicle.

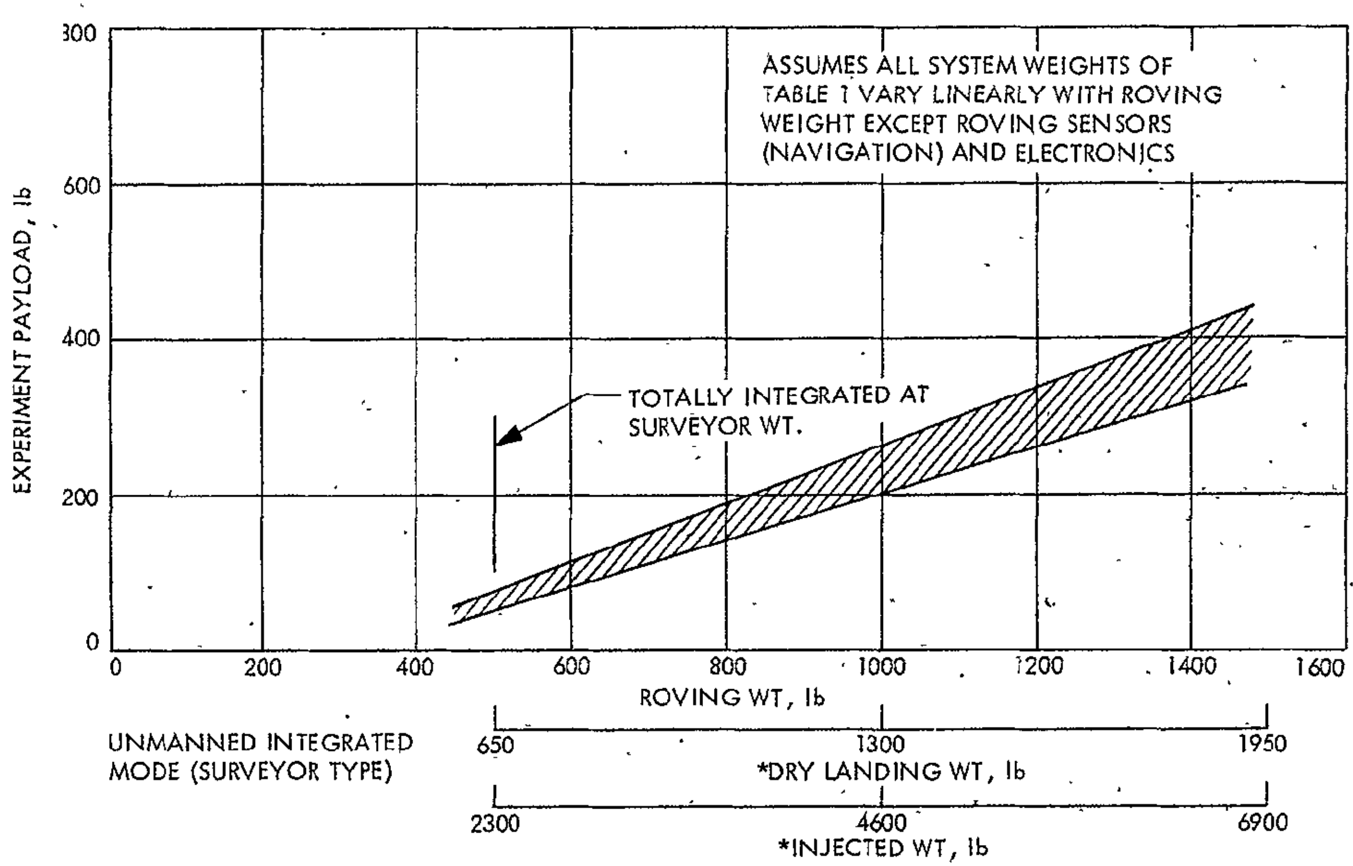

Fig. 10. Payload weight as a function of roving vehicle weight

2. Flight profile. Although this concept is considerably different from the Surveyor in configuration, the general sequence of events during flight and lunar descent of these two vehicles is essentially the same. The various phases of operation are as follows:

During launch the spacecraft is contained within the Atlas-Centaur shroud and supported on the Centaur tank dome in the normal Surveyor manner (See Fig. 9, Boost Configuration). In this phase the solar panel, antennae and wheels are in their stowed position. After standard shroud ejection and just prior to spacecraft separation the solar panels and antennae are erected and the wheels spring to their roving position. Events after separation through midcourse and to lunar arrival are then identical to the Surveyor with the configuration as shown on Fig. 9, Cruise Configuration.

The descent phase is essentially the same as for the Surveyor in that the main retro and vernier motor plus the AMR and RADVS descent systems provide the standard controlled descent to a soft landing on the lunar surface. (See Fig. 9, Cruise and Landed Configuration.) However, a variation is introduced into the final touchdown phase just prior to surface contact for the purpose of softening the landing impact. This is accomplished by continuing the constant 5 ft/sec final Surveyor descent velocity (controlled by the vernier motors) to contact. Motor cutoff is then actuated by a surface contact switch (similar to that previously considered for both Apollo and Surveyor). This variation, which eliminates the 18 ft (maximum) free fall period of the Surveyor, minimizes the rover wheel and carriage weight penalties due to landing impact loads. At the same time this eliminates the need for the cold gas attitude control system for the free-fall descent and permits its early jettisoning along with the AMR.

3. Vehicle performance. There are two possible configurations being considered for the roving phase. One is the as landed condition (Fig. 9, Landed Configuration). Here the descent systems are retained during roving, and could be used for emergency 'hops' in the fashion planned to be tested on Surveyor. This results in a greater traverse weight, less obstacle clearance, and an increase in traverse power as compared to the minimum weight configuration.

The other configuration, having a smaller traverse weight, involves the jettisoning of the expended descent system after landing. This is accomplished by releasing the hinged and spring attached "A" frame which flips off to one side. The purpose of this operation is to minimize the roving weight and reduce power requirements.

This roving vehicle is powered by an electrically driven motor in each wheel. Center point steering is also electrical; motion and steering are ground commanded. Imaging systems, including stereo TV and a ranging sensor, provide navigation information for directing the vehicle along its route. To limit demands on the control system and the Earth-based operators, internal hazard-sensing devices (e. g. tilt and obstacle detectors) are provided. Operation of the vehicle during roving is intermittent in that power is first supplied to travel a short distance (approximately 1,000 ft) and then stop. Power is then supplied to perform a new visual survey and where appropriate to make a series of experiments and to transmit the information to the receiving stations. This intermittent roving and experimenting sequence is then repeated.

In keeping with the philosophy of using as much Surveyor hardware as possible the baseline design has a "Dry Landed Weight" of:

642 pounds Dry Landed Weight for
66 hour nominal flight
30 meter per second midcourse
25° or less approach angle
2670 meters per second approach velocity
1300 pound retro propellent
177 pounds useable vernier fuel

or

660 pounds Dry Landed Weight for
90 hour nominal flight
30 meter per second midcourse
25° or less approach angle
2540 meters per second approach velocity
1300 pound retro propellant
182 pounds useable vernier fuel

The following weight estimate is based on using known Surveyor equipment where applicable and on estimates for the remaining equipment (Table 3).

Table 3. Weight breakdown for lunar rover

Characteristics	Flight configuration		Descent and land configuration	Travel configuration
Navigation Equipment		50 00	50.00	50.00
Flight Control		57.33	17.00	
Data Link		35 07	35 07	35.07
Central Command Decoder		5.73	5 73	5.73
Central Signal Processor		5.93	5.93	5.93
RADVS		34 06	34.06	
Boost Regulator		8.22	8.22	8.22
Engineering Signal Processor		5.99	5.99	5 99
Engineering Mechanisms Auxiliary		4.70	4 70	
Battery Charge Regulator		3.47	3 47	3 47
Power Switch		1.00	1.00	1 00
Electrical Power		106.40	106 40	106 40
Battery	46 40			
Solar Panel	30 00			
RTG	30 00			
Mechanisms		18 05	18 05	18.05
Pin Pullers	2.00			
Separate Sensor and Arming	1.05			
Positioners	15.00			
Basic Vehicle		449 49		
Basic Structure	50 00		45.00	40 00
Equipment Attach Hardware	20 00		18.00	10 00
Paint	1 00		1.00	0 60
Landing and Traverse Gear	85.00		85.00	85.00
Wiring	25.00		23.00	20 00
A/C Lines	1.00		1.00	
Retro Release Mechanism	1 80		1 80	
Centaur/Spacecraft Latch	0.66		0 66	
Vernier Propellant System	80.84		80.84	
Vernier Propellants	182 07		162 00	
Unuseable Vernier Propellants	2 12		2.12	
"A" Frame	15.00		15.00	
Contingency		30.00	30.00	30 00
Retro-Rocket		1442.94	1442.94	
Case, Insulation, Release, Wiring	142.94			
Propellant	1300.00			
Altitude Marking Radar		11.74	11 74	
Buoyancy		1.00	0.90	0 50
Total Weight Less Payload (90 hr trajectory)		2286 12	2216 62	425 96
Expendables				
Nitrogen Gas	4.50		1627.73	
Altitude Marking Radar	11.74			
Helium Gas	2.41			
Useable Vernier Fuel	162.52			
Unuseable Vernier Fuel	2 12			
Retro Case	142 94			
Retro Propellant	1300.00			
Ballast-Retro	1.00			
Buoyancy	0 50			
Dry Landed Weight Less Payload			588.89	
Maximum Dry Landed Weight (90 hr trajectory)			660.00	
Maximum Payload (90 hr trajectory)		71.11	71.11	71.11
LRV Traverse Gross Weight (90 hr trajectory)				497 07
Total Separated Weight at Injection (90 hr trajectory)	2357.23			

*Weights are preliminary, decimals are carried in table merely to permit entry of actual Surveyor item weights where appropriate.

Drawing on the power requirements for the known Surveyor equipment and estimating the power requirements for the new equipment, the LRV System Peak power demands for the various mission phases are:

Stabilization, Sun Orientation and Canopus seeking	180 w. reg.
First Cruise	110 w. reg.
High Power Interrogation	180 a. reg.
Mid Course Maneuver	310 w. reg.
Second Cruise	110 w. reg.
High Power Interrogation	180 w. reg.
Terminal Maneuver and RADVS	890 w. reg.
Traverse	150 w. reg.
Traverse Stall Power	290 w. reg.
Experiment Operation & Communication	150 w. reg.

All LRV operations will be initiated by ground command except those which are so time constrained, by elapsed time or time sequencing accuracies, that they must be initiated by on board automatic means. All automatic operations will be capable of being backed up by ground commands whenever possible. The current Surveyor, command system uses 256 commands having a hamming distance of 4 between them to avoid acting on errors in sent commands. It would require a 4 bit error to put a wrong command into the LRV.

4. Guidance and Navigation. All LRV operations will be solely ground commanded except for:

(a) Terminal retro and landing sequence.

(b) Automatic sun acquisition at injection.

(c) Obstacle avoidance or other hazard alarm.

(d) Maintenance of Earth view.

The LRV equipment for guidance, control and navigation falls into two categories.

(1) Flight Control.

(2) Lunar Traverse.

The flight equipment is, for this study, considered to be the same as that used on Surveyor. The flight control subsystem controls the LRV velocity and attitude during the transit phase of the mission. This phase covers the period from separation

of the LRV from the Centaur vehicle to touchdown on the lunar surface. The basic functions performed by the flight control subystem include: (1) attitude stabilization and orientation during the entire transit phase, (2) midcourse trajectory correction based on radio command data, and (3) terminal phase retro maneuver and vernier descent for landing of the LRV in an upright position on the lunar surface. Three principal forms of reference — celestial reference, inertial reference, and RADVS and AMR control — are used.

The flight control sensor group, which is composed of a group of optical and inertial sensors and the flight control electronics, controls LRV velocity and attitude during transit. This is accomplished by providing thrust commands to the vernier engines, main retro engine, and cold gas jets, and position commands to the roll actuator.

The specific functions performed by the flight control sensor group include (a) sensing angular velocity for attitude stabilization, (b) control of a sequence of LRV maneuvers for initial sun and Canopus acquisition, (c) generating error information from the primary sun sensor and Canopus sensor to slave the spacecraft attitude to the sun and Canopus references, (d) providing angular information to maintain an inertial reference during midcourse velocity correction and the initial stage of the terminal descent maneuver, (e) supplying angular memory for precision attitude maneuvers, (f) providing precision acceleration control for midcourse velocity correction, and (g) processing range and velocity information from the radar altimeter and doppler velocity sensor to generate thrust commands for landing. The flight control sensor group is composed of the following subassemblies: (a) the inertial reference unit for inertial sensing, (b) automatic sun acquisition sensor, primary sun sensor, and Canopus sensor for optical sensing, (c) retro accelerometer and inertia switch for acceleration sensing, (d) control circuits and (e) timing and mode selection.

After landing, remote guidance of an unmanned lunar roving vehicle can be accomplished in a variety of ways. One method being considered, based on a minimum of on-board navigation equipment, involves a sequence of short discrete commanded "steps". Each step consists of a position-determining-and-decision period and an automatic traverse period. Navigation aids will include but are not necessarily limited to one or more inclinometers, one or more accelerometers and an optical

ranging device. In addition there will be automatic obstacle avoidance equipment and equipment to assure that the LRV is in view of the earth in order to maintain the communication link.

The LRV wheels will be driven by electric motors and the surface speed of the vehicle will be evaluated based on the following mission requirements:

(1) The vehicle lifetime will be one year.

(2) The sum of the great circle distances travelled by the rover during its lifetime will be approximately 1500 km.

(3) The vehicle will be "driven" by operators at one location. Prime operation will be via Goldstone but limited operation through the other Deep Space Net stations will be possible. The normal operating mode will consist of (a) determining vehicle location and orientation by visual inspection, (b) orienting the rover, (c) commanding a traverse of distance W, and (d) monitoring engineering telemetry until the vehicle signals it has stopped again. This procedure, repeated throughout the rover's journey, may be altered for short periods, as when the traverse path contains numerous obstacles or during investigation of interesting areas.

(4) The vehicle will be capable of detecting obstacles and hazards but will not automatically avoid them. Normal operation upon detection of a dangerous situation will be to stop and wait for operator intervention.

(5) Navigation aids carried on the vehicle will be a simple ranging device such as a laser (range about 100-1500 m.), one or more odometers, one or more inclinometers, and a real-time panoramic stereo/monoscopic imaging system.

The feasibility of any traverse capability rests mainly upon a reasonable average speed for the vehicle. To derive the expected average speed one must first anticipate the amount of time spent actually moving the vehicle closer to its ultimate destination.

A number of factors influence the allowable average vehicle speed. Several of them reflect in an increase in the actual distance traversed. The remainder serve to decrease the actual traverse time.

The factors affecting traverse distance (and their assumed magnitudes) are:
Non-spherical surface (5%)
Terrain ruggedness (10%)
Navigation and guidance errors (10%)

Feature investigation (5%)
Obstacle avoidance (5%)

The factors affecting traverse time are:
Lunar day travel only (45%)
Goldstone operation (33%)
Remainder DSN operation (50%)
Feature investigation (5%)
Obstacle encounters (10%)

Non-spherical surface refers to relatively slow altitude variations while terrain ruggedness involves more abrupt changes such as craters, mounds etc. Navigation and guidance errors will result as the LRV departs from a straight path during a traverse. At the end of each "step" it will probably be displaced from the desired destination. This displacement will have to be corrected in the succeding step(s). Feature investigation accounts for the extra movement and "poking around" required at each sample-collection site.

Turning now to the factors affecting traverse time, it will probably turn out that some traverse will be possible at night if solar-independent power is available. However, with traditional conservatism and because the availability of a large solar-independent power source is not assured, we have anticipated daytime travel only. It was further assumed that travel would be inhibited, or at least difficult, when the solar elevation was less than 10° or when the LRV was "looking" into the sun.

Operation via Goldstone is possible of course for only approximately one-third the time. It was assumed that operator capability will be slowed by 50% for the remaining two-thirds time when the other DSN stations are in view of the LRV. This is a reasonable assumption particularly if high-resolution pictures must be relayed to the central control point in the U.S.

The factor for feature investigation allows for time (other than that required for rover travel) for operators' and scientific investigators' decisions at sampling localities. The factor introduced for obstacle encounters covers expected periods of operator familarization and decision whenever a signal is received that a hazard has been encountered.

Final values for traverse distance and time can now be obtained. Traverse distance is found from:

$$\text{Traverse distance} = D = 1500 + 1500\,(.05+.10+.10+.05+.05)$$
$$= 1500(1.35) = 2025 \text{ km.}$$

The time factors are combined differently. First the total available lunar day time is found:

$$\text{Available lunar day time} = T_d = .45\,(8700) = 3920 \text{ hr.}$$

The DSN limitations and other time elements can then be factored in:

$$\text{Traverse time} = T = \left[.33\,T_d + .5\,(.67)\,T_d\right]\,(1 - .05 - .05)$$
$$= .9\,(.67)\,T_d = .6(3920) = 2350 \text{ hr.}$$

The total traverse time is divided into periods of movement M and stationary S periods during which operator decisions are being made. The majority of the LRV active lifetime will consist of alternating periods of M and S. The final step in obtaining the LRV average speed is to determine the average values of these two quantities.

The following activities will probably typify one S period:

1. Determination of vehicle position and orientation.

(a) A quick monoscopic sweep of the area.

(b) Move vehicle to nearby level surface (or level camera base).
Detailed horizon sweep with operator examination of features. Assume 5-10 pictures. If pictures are monoscopic they may be viewed simultaneously on a number of screens. If they are stereoscopic they will be viewed sequentially and the operator will be able to recall any stereo pair for additional study.

(d) Locate observable features on lunar maps.

(e) Range to at least three features. Aiming direction of the ranging device will be displayed on the operators' viewing screens. Aiming accuracy will be 1.0° and ranging accuracy will be ten meters.

(f) Triangulate on lunar maps. Laying off lines in the ranging directions and marking distances, the operator can determine with acceptably small error where the LRV is and the direction it is facing.

2. Tactical decision for next step.

(a) Choose traverse path to correct deviations from planned path. Draw new path on map.

(b) Stereoscopic examination of new path. Look for obvious dangers and adjust path accordingly.

(c) Choose step distance as a function of path character and past operator experience.

(d) Reorient LRV in direction of path.

(e) Command traverse.

Time durations are assigned to each of the above activities. These are average times. Obviously, initial operator efforts will be slower. Of course, if the LRV winds up in the bottom of a crater when it stops, the operator will require longer than usual to start the next step. There will probably be times when no identifiable features can be seen and the direction of shadows and the Earth will have to provide clues to the vehicle's orientation and traverse path. But there will also be times when features are immediately picked out on the first TV sweep and path deviation is negligible. Furthermore, the time durations indicated below are based on the assumption that the step size is short enough so that the LRV terminates a traverse at a point that has been visible from the previous location. Thus features should be easy to identify. Therefore, we assert that the times stated are reasonable averages over the entire traverse. Later it is shown that the step size is indeed short enough for this conclusion.

	ACTIVITY	DURATION (minutes)
1a.	area sweep	1/2
b.	leveling	3/4
c.	detailed view	2
d.	feature indent.	1/2
e.	ranging	1
f.	triangulation	4
2a.	new path choice	1
b.	new path exam.	1
c.	step dist. choice	1

	ACTIVITY		DURATION (minutes)
d.	vehicle orient.		1/2
e.	traverse command		1
	total	S =	13-1/4

With p as the total number of steps we can write

$$T = p\ (M + S)$$

$$\text{and } D = pW = pMV_A$$

where W is the average step size and V_A is the average LRV velocity. Combining the two equations results in

$$D = TV_A \frac{M}{M+S}$$

With D, T and S known, V_A may be plotted as a function of M. The graph is shown in Fig. 11. Numbers at the plotted points are the corresponding calculated values for W, the step size. Every point on the curve satisfies the above equations. Any reasonable combination of V_A, M and W may now be chosen to justify the previously stated assumptions. For instance,

V_A = Average speed = 1.25 km/hr. = .684 ft/sec.

M = Average step time = 0.5 hr.

W = Average step distance = .625 km. = 2050 ft.

5. Rocket propulsion. The propulsion subsystem components supply the thrust forces for maneuvering the spacecraft during the midcourse correction and lunar landing phases of the mission.

The propulsion subsystem consists of three liquid-fueled, variable-thrust, vernier engines for midcourse velocity vector correction and landing phase maneuvering, and a solid propellant retro rocket engine for supplying the principal deceleration force during the landing maneuver. Each portion of the propulsion system is controlled by the flight control system through pre-programmed maneuvers, commands from earth, and maneuvers initiated by flight control sensor signals.

Vernier propulsion supplies the thrust forces for midcourse maneuver velocity vector correction, attitude control during retro rocket engine burning, and velocity vector and attitude control during terminal descent to the lunar surface. The vernier engine system consists of three thrust chamber assemblies and a propellant feed system. The feed system is composed of a fuel tank, an oxidizer tank, a

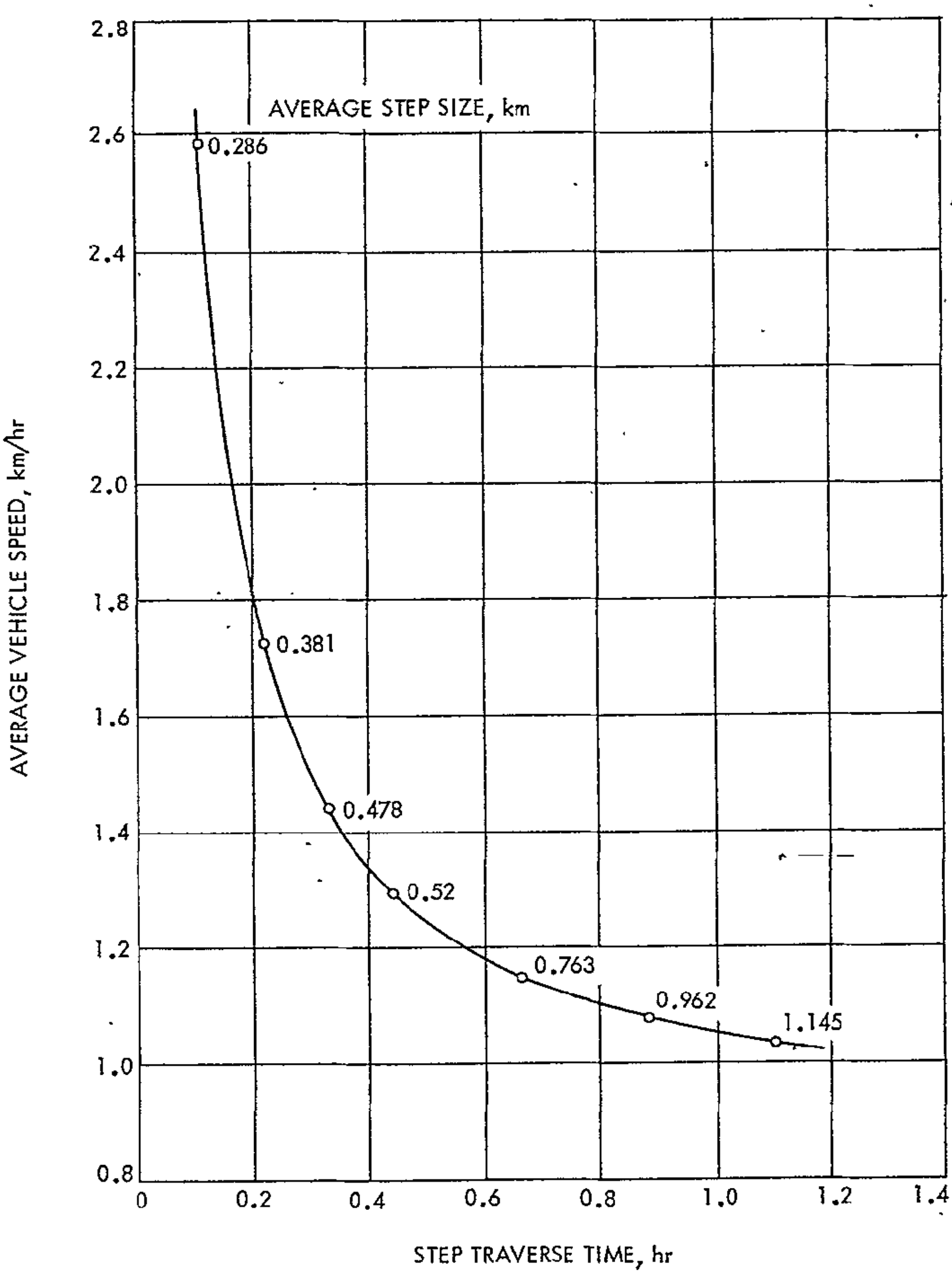

Fig. 11. Average roving speed as function of go-stop step size, with allowances for imaging and navigation.

high-pressure helium tank, propellant lines, and the necessary valves for system arming, operation, and deactivation. The arrangement of the tanks on the spaceframe is illustrated in Fig. 9. Tanks, and some segments of the propellant lines, are electrically heated to maintain the propellant temperature above 0° F. Thermal sensors on all tanks, all engines, the helium tank, and the three propellant line segments permit telemetering thermal data. Fuel and oxidizer tanks each contain positive expulsion bladders which deflate around a central standpipe to permit complete expulsion and assure propellant delivery under zero-g conditions. Helium release and dump valves are squib-actuated upon command. The helium tank stores gas under pressure to force the propellants into the thrust chambers. Valves permit release of helium to the system, regulation of pressure, and dumping of residual helium.

The thrust chambers are located on the "A" frame on the top of the main structure. One engine is hinge (swivel) mounted on an electro-mechanical roll actuator motor to rotate the engine about an axis in the spacecraft x-y plane for roll control. The roll actuator positioning lock is released subsequent to boost by a squib-actuated pin puller. The LRV control moment arm of each engine is approximately 38 inches in length. The specific impulse and total impulse vary with engine thrust.

The thrust of each engine can be throttled over a range of approximately 30 to 104 pounds by a bipropellant throttling valve using control signals supplied by flight control. Engine firing is accomplished by solenoid-controlled, helium-actuated on/off valves controlled by the flight control electronics. Throttling valves are controlled individually while the on/off valves operate collectively from a single signal. The oxidizer is nitrogen tetroxide (N_2O_4) with 10 percent by volume nitric oxide (NO) to depress the freezing point. The fuel is monomenthyl hydrazine monohydrate ($72\ N_2H_3(CH_3) \cdot 28\ H_2O$). Fuel and oxidizer ignite hypergolically in the thrust chamber.

The thermal control design of the vernier engine and feed system maintains the temperature of all portions of the system between 0 and 100 F during non-thrust periods (from launch to touchdown) preventing propellant freezing or overheating by a combination of active and passive thermal controls utilizing surface coatings and electrical heating. Other system components are thermally isolated from the structure to ensure that the LRV structure acts as neither a heat source nor heat sink.

The main retro rocket, which performs the major portion of the deceleration of the LRV during lunar landing maneuver, is a spherical, solid-propellant unit with a partially submerged nozzle.

The unit is attached at three points on the main structure with explosive nut disconnects for post-firing ejection. Friction clips around the nozzle flange provide attachment points for the altitude marking radar and portions of the flight control system not used after retro-ignition. Retro rocket igniter gas pressure ejects the altitude marking radar and flight control equipment when the retro firing sequence is initiated. Retro rocket ignition squibs and retro release explosive nuts operate from a pulsed 19-ampere constant-current source. Commands are initiated by the flight control system.

The retro rocket safety and arming device has dual firing single-bridgewire squibs for firing the retro rocket igniter. In addition, provisions for local and remote safe and arm actuation and remote indication of inadvertent firing of the squibs are included. Both mechanical and electrical isolation exists between squib initiator and pyrogen igniter in the safe condition.

The retro rocket with propellant weighs approximately 1443 pounds. The engine thrust may vary from 8000 to 10,000 pounds over the temperature range of 50° to 70° F.

Two thermal sensors are installed for monitoring retro rocket nozzle temperature before ignition.

The thermal control design of the retro rocket engine is completely passive, depending on its own thermal capacity, insulating blankets, and surface coatings to maintain the "cold spot" propellant temperature above 17° F at the time of ignition. Because of the thermal gradient through the engine and the prelaunch engine temperature, the 17° temperature will be reached at the three engine attachment points only. The bulk temperature of the propellant grain will be above +50° F.

The engine utilizes an aluminum, ammonium perchlorate, polyhydrocarbon, case-bonded composite-type propellant and conventional grain geometry. The nozzle has a graphite throat and a laminated plastic exit cone. The case is of high-strength steel insulated with an asbestos and silicon-dioxide-filled buna-N rubber to maintain the case at a low temperature level during burning.

6. Telecommunications Subsystem. The telecommunications subsystem consists of three interconnected groups which provide command reception and decoding, telemetry signal processing, transmission, two-way coherent doppler tracking and LRV to Earth base range determination. These groups are: (1) a data link group that provides RF transmission and reception, (2) a command decoding group that provides decoding logic functions for all Earth commands, and (3) a signal processing group that provides commutation, analog-to-digital conversion, and modulation functions for processing of analog, digital, and video (T.V.) data channels. A telecommunications system block diagram is presented in Figure 12.

The data link group provides a two-way communication link between the LRV and DSIF tracking stations throughout the mission. The data link group receives command signals transmitted from DSIF stations to the LRV and transmits signal data from the LRV to DSIF tracking stations during launch, injection, transit, landing and lunar operations. The data link group consists of a transponder (RF receiver and RF transmitter), a command decoder, an engineering and scientific signal processor, a central signal processor, and telecommunications antennas.

The S-band transponder provides the functions of a double superheterodyne command receiver, a phase-coherent transponder, a turnaround ranging transponder, and a telemetry transmitter. It employs three loops, the Ranging, Automatic Gain Control (AGC), and Automatic Phase Control (APC) loops. In the absence of a received signal from the DSIF, the spacecraft telemetry information is modulated on an RF signal whose frequency is controlled by an auxiliary oscillator. When a signal from the DSIF is present in the transponder receiver, the APC loop will lock to it, and AGC voltage is developed in the AGC loop. This voltage, besides controlling the receiver gain for proper operation, also produces a command voltage which turns off the auxiliary oscillator and causes the VCO to be switched to the transmitter input in its place. In this manner, closing the APC loop provides an exact ratio of 240/221 between the transmitted and received frequencies. In addition, the AGC voltage is telemetered to provide received signal strength information.

The signal received from the DSIF may have either ranging information or command information phase modulated on it. The second mixer provides two outputs, one to the narrow-band command and APC channel, and another to the broad-band ranging channel. The ranging code is demodulated, filtered, limited, and remodulated onto the transmitted signal along with the telemetry. The command information is demodulated by the phase detector in the narrow-band channel.

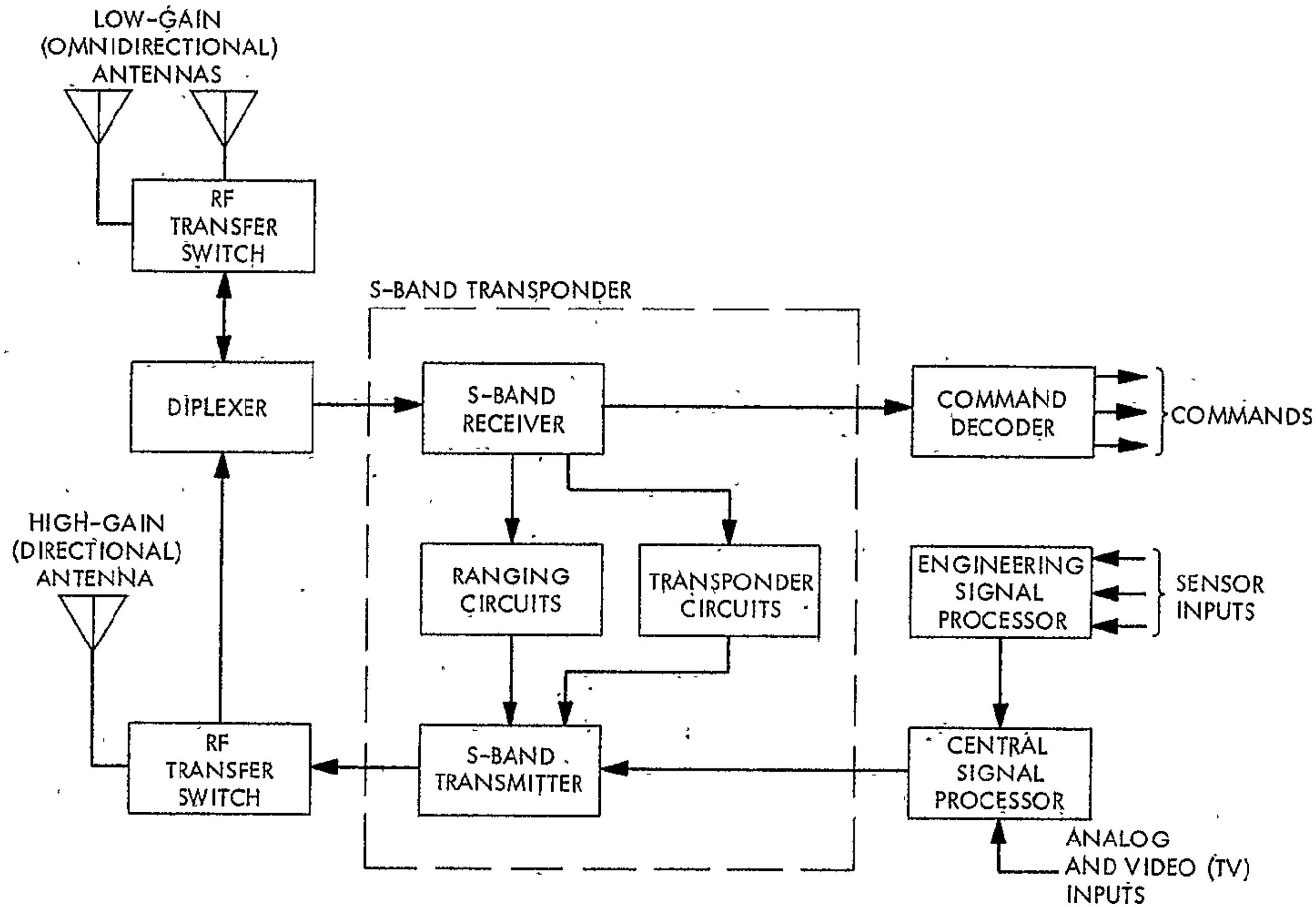

Fig. 12. Rover telecommunications subsystem

The transponder considered above is of a unique class in that it provides for a turnaround ranging capability that is compatible with the NASA-DSN tracking stations. This type of ranging transponder has been used previously on such programs as Mariner, Lunar Orbiter, and Apollo. The Surveyor transponder was not considered for the LRV since it does not have ranging capabilities.

Three telecommunications antennas are provided on the LRV; two are omni-directional antennas for command reception and transponder operation, and the third is a high-gain, directional antenna used to radiate sufficient effective power for real-time television transmission. An antenna switching function is included to allow use of alternate antennas for transmitting and receiving.

The basic functions of the LRV command system are:

(1) Accept a command subcarrier from the LRV receiver.

(2) Demodulate the command subcarrier and receive the digital command information.

(3) Provide timing, decoding of the digital command information and error detection.

(4) Issue commands in a form suitable for execution by the appropriate LRV subsystems.

The nature of the mission to be performed by the LRV is such that it requires initial control of each step by command from Earth. The LRV Command Subsystem will be exercised to a great extent throughout the vehicle lifetime. In choosing a command subsystem it is therefore important to make sure that its operation will not place any limitations on the performance of the mission. The command parameters that are most important to the LRV operation are the command word time, the command word error rate, and the operational complexity involved in sending a command.

Since the mission consists of many discrete steps and each step may require several commands, it is important to keep the command time short to minimize mission time. It is anticipated that no provisions will be made for the verification of correct command reception through the telemetry system prior to command execution. Because of Earth-Moon propagation times and decommutation delays such verification would add significantly to mission time. It is therefore important that a command be received correctly on the first try, both to minimize mission time and to increase

operation confidence. Finally, because the command function is to be exercised so often the procedure for sending commands must be made as simple as possible.

An estimate of a satisfactory upper bound on the command word time can be obtained by the following argument.

The approximate time required to get a command from the mission operations control center (MOCC) into the LRV's command decoder is the command word time plus the MOCC-Moon propagation time. A conservative estimate of the MOCC-Moon propagation time is 3.5 seconds with a breakdown shown below.

(a)	Transmission to remote tracking station by ground communication circuits.	2.0 seconds (a) and (b)
(b)	Error checking, reformatting and transmission to LRV	
(c)	Earth-Moon propagation time	1.25 seconds
(d)	LRV decoding time	0.25 seconds
		3.5 seconds

To minimize the overall command transmission time, it is desirable to make the command word time short compared to 3.5 seconds.

This is quite possible with a command channel bit rate of 100 bps, such as the current Surveyor design. Assuming a library of 256 commands, a command word must contain 8 information bits. Assuming 6 more bits for word sync and parity checks, the total command word becomes 14 bits long. At 100 bps this translates into a word time of about 0.14 seconds, which is short compared to 3.5 seconds.

The signal processing group gathers the engineering and verification signals from various subsystems and provides the appropriate signal conditioning. The signal processing group comprises the engineering signal processor and a central signal processor.

The engineering signal processor processes data from the LRV and in conjunction with the central signal processor, puts it in a suitable form preparatory to being transmitted to Earth. This processor handles all data required to assess the performance of the basic bus (engineering data).

The engineering signal processor is composed of the following major components: commutators, current sources for thermal measurements, a command enable and reject channel, accelerometer channel (if required), and a gyro speed channel for the inertial reference subsystem that is required during transit between the Earth and the lunar surface.

The engineering signal processor contains a commutator capable of several modes of operation for priority engineering data as a function of mission phase (i.e., launch, transit maneuvers and cruising, and lunar roving).

The central signal processor combines the outputs of the engineering signal processor and other units. These signals are processed and sent to the frequency or phase modulation inputs of the transmitter. The subsystem has the capacity for handling analog data and converting it to digital data for subsequent transmission. Synchronizing patterns, parity digits, and timing signals for controlling commutators are generated in the central signal processor.

The design of the telecommunications subsystem for a Lunar Roving Vehicle involves the identification and analysis of several tradeoffs. The most fundamental to the subsystem design are the tradeoffs which affect the gain-power product. Since it is anticipated that the most severe communications requirement will be for high resolution television, the tradeoff between television quality and gain-power product becomes important. The term "television quality" refers to the two parameters of image resolution and frame transmission time. Another tradeoff of interest is antenna size versus the problem of pointing the antenna at the receiving DSIF station. This problem is compounded as the LRV mission moves away from the sub-Earth point.

The need to obtain engineering and science telemetry and television from the LRV is obvious. It is also desirable to obtain the telemetry data or television frame in a minimum length of time, or in other words, transmit from the rover to Earth using a high data rate (or bandwidth). The cost of a high data rate is measured in terms of the amount of transmitting power and antenna gain required. This is referred to quantitatively as gain-power product and is usually expressed in decibels referenced to 1 milliwatt (dbm). The cost, or disadvantages, of a high-gain antenna are essentially greater weight and more antenna pointing problems. The cost of high transmitter power is measured in terms of the weight of the raw power source required to drive it. If we assume high power transmission only when the LRV is at rest, the locomotion power becomes available for the transmitter.

Figures 13 and 14 are plots of required gain-power product versus telemetry bit rate and communications base bandwidth.

An imaging system is required on the rover as a navigation aid. It is also a science objective to obtain high resolution pictures of the lunar surface. To meet these requirements the communication system must be capable of transmitting the video data to Earth in a relatively short time so as not to impair the overall efficiency of the LRV.

The speed with which a television picture can be transmitted from the Moon to Earth is a function of the picture resolution and the available bandwidth. Maximum available bandwidth is a function of the LRV's gain-power product and is shown in Figure 14 for a typical set of link parameters.

Figure 15 is a plot of television frame transmission time versus gain-power product for several assumed resolutions. Assumptions include phase modulation at the optimum modulation index with narrow band phase-lock loop detection on the ground. Video band signal to noise is 36 db peak-to-peak signal to RMS noise. Use of an 85 foot tracking antenna with a maser front end is assumed. An arbitrary margin of 6 db was selected. If a 210 foot tracking antenna is used, the curves of Figure 15 would be shifted about 8 db to the left, reducing transmission time by a factor of 6.

It should be noted that for transmission of a stereo pair, transmission time would be doubled.

In order to satisfy the mission requirements and constraints there are 3 antenna configurations that must be considered. These are: (1) omnidirectional, (2) directional, and (3) a combination of omnidirectional and directional.

Some of the mission requirements (e.g., command reception, telemetry, and ranging) make an omnidirectional antenna configuration appear to be the most desirable. However, other mission constraints and requirements (e.g., LRV operation over the Earthside hemisphere of the Moon and a direct LRV communications link with Earth-based tracking stations) coupled with the present capabilities of telecommunications hardware (e.g., available RF transmitter power) make a directional antenna configuration mandatory.

Therefore, it is concluded that a combination of omnidirectional and directional antenna configurations will satisfy the mission requirements and constraints

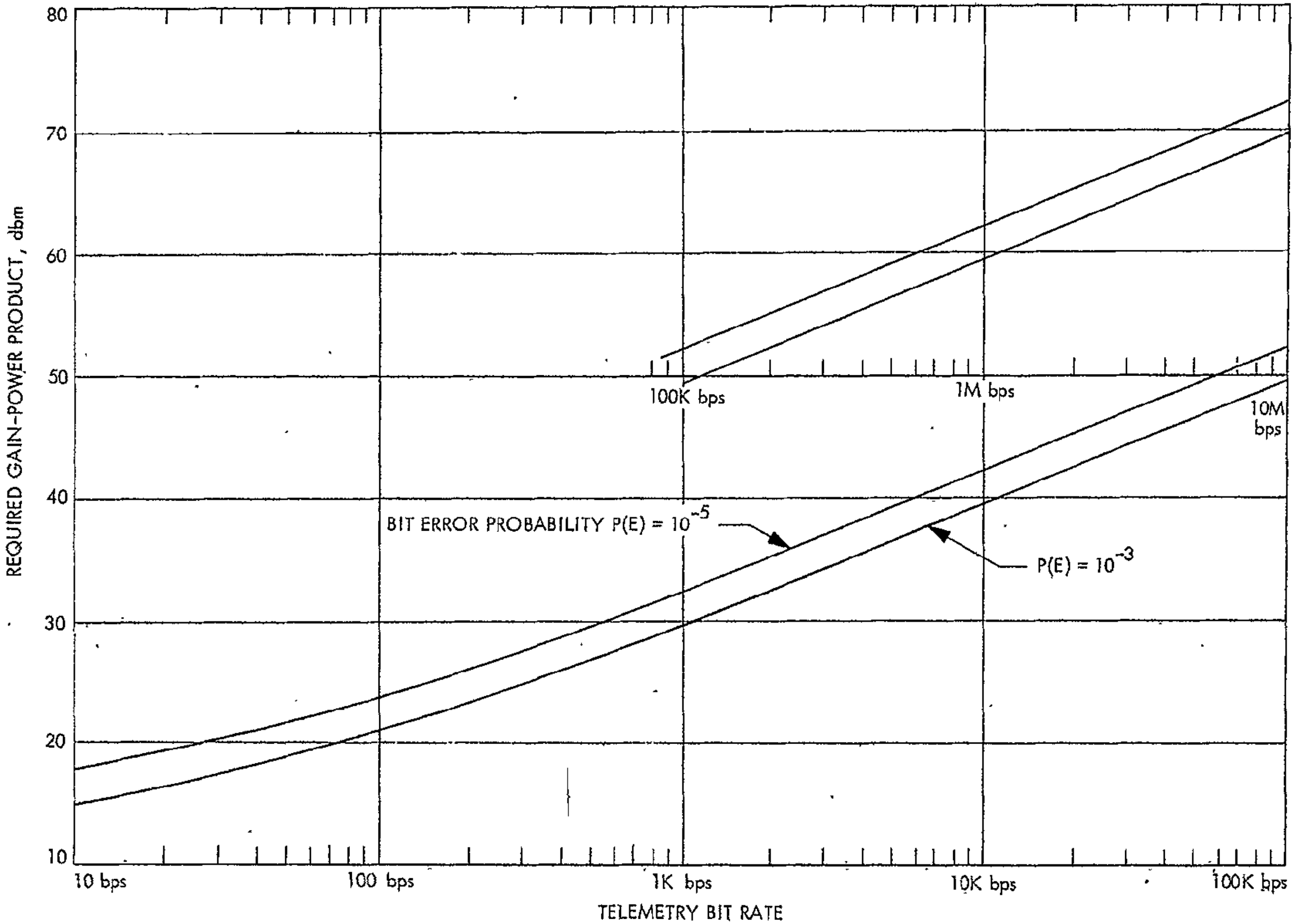

Fig. 13. Required gain-power product vs bit rate

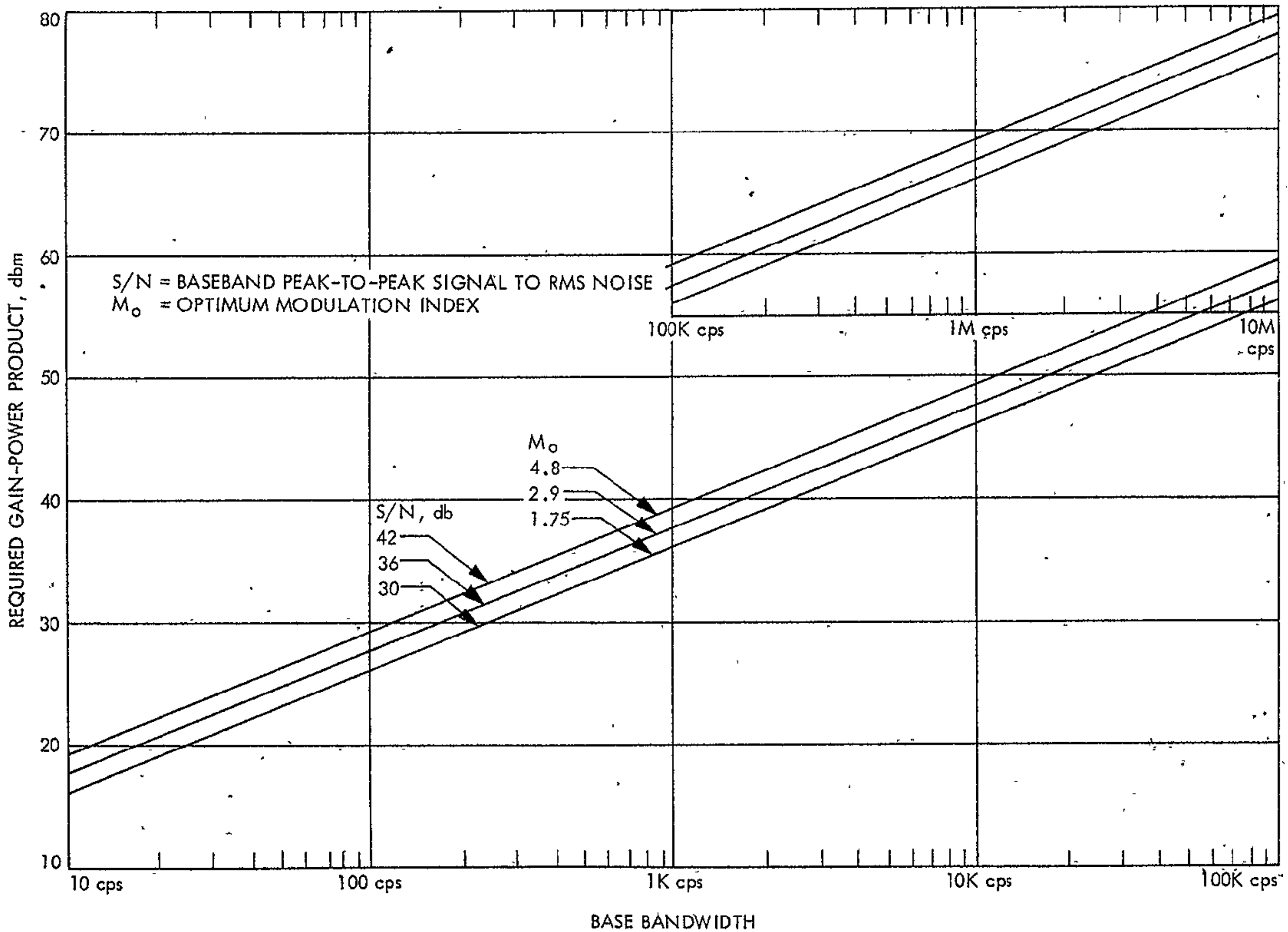

Fig. 14. Required gain-power product vs bandwidth

760-5

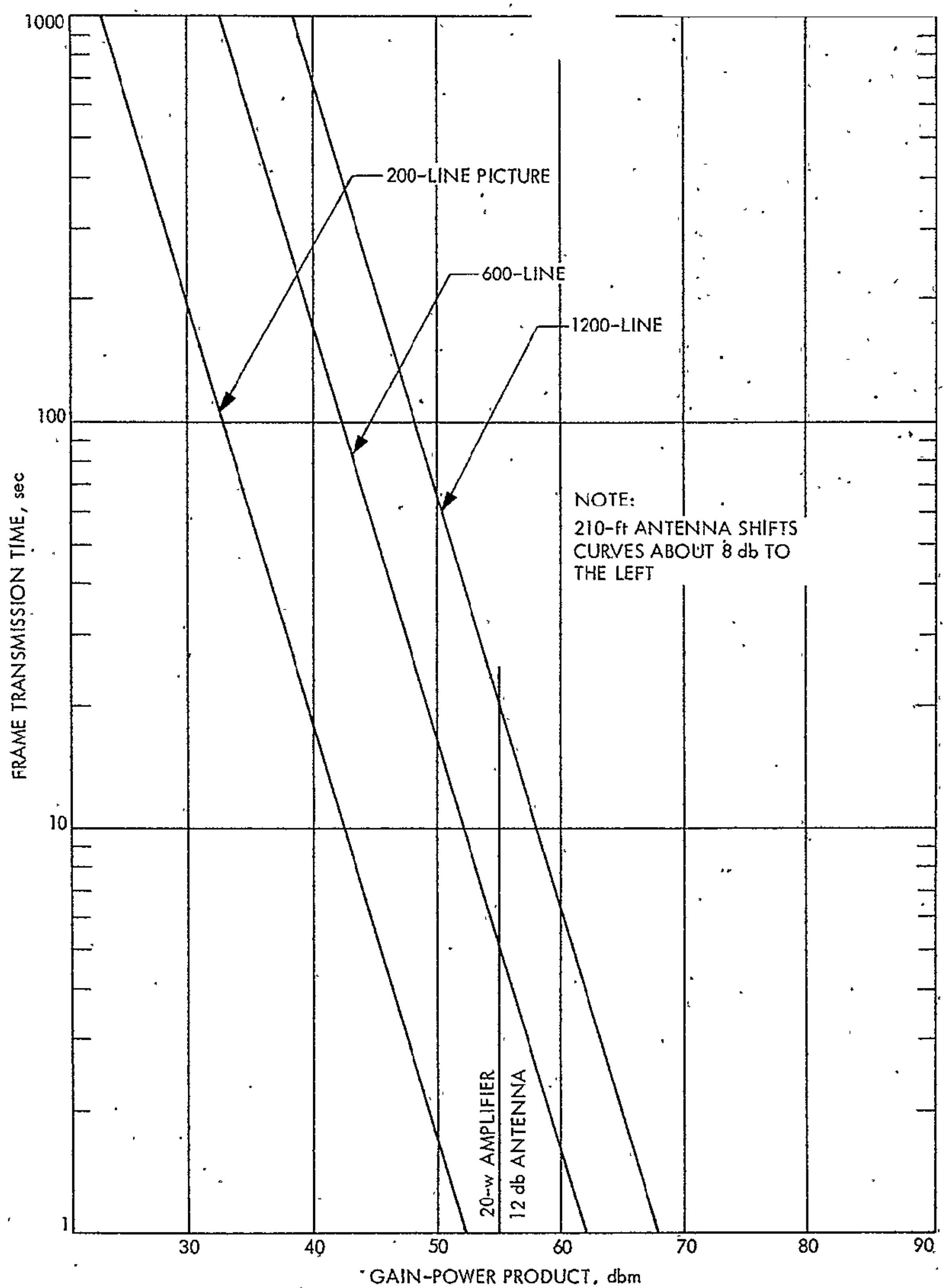

Fig. 15. Frame time vs gain-power product

To obtain a high reliability in the antenna system, the omnidirectional antenna should be rigidly mounted to the LRV. The present Surveyor omniantenna concept is adequate for the LRV application. The Surveyor antenna has a radiation pattern which is relatively uniform over a hemisphere. If the antenna were placed on the LRV such that the RF axis is perpendicular to the lunar surface, one antenna would be satisfactory. However, if the RF axis is parallel to the lunar surface, two omni-antennas would be needed as presently used on the Surveyor spacecraft.

Hemispherical coverage by a directional antenna necessitates that the directional beam must have some form of steering. Three classes of steering are feasible: (1) electronic steering, (2) mechanical steering, or (3) a combination of electronic and mechanical steering. Each class of steering may be divided into two basic types: (1) closed-loop, or automatic steering, and (2) open-loop steering. Due to the complexity of the LRV mission, the open-loop steering system appears to be a likely choice. The open-loop system would require no Earth sensing system on the LRV, thereby reducing the antenna complexity, weight and power.

For an open-loop pointing system, assume the following model for an analysis of a simple antenna pointing scheme. The LRV moves from point A to point B by several discrete steps. At the end of each step, the following activities are performed to prepare for video transmissions:

(a) Rover heading and attitude are sensed by the navigation system sensors and the readings telemetered to Earth.

(b) Data from (a), along with other necessary data, are fed to a ground computer which calculates the correct antenna position.

(c) Antenna position commands are encoded and transmitted to the rover.

(d) Commands are decoded by the rover and the antenna is repositioned.

The total time needed to point the antenna is the sum of the times to complete the four functions described above. The average or expected value of the antenna pointing time increases as the antenna gain increases. As the antenna gain increases, the beamwidth decreases, and the probability that a DSIF tracking station is within the antenna beamwidth at the end of a traverse step decreases. Therefore, for higher antenna gains it is more likely that the antenna will need repositioning which involves steps (c) and (d) above. This argument assumes that when a DSIF tracking station is within the beamwidth, no positioning is required and the pointing operation

s completed after step (b). Figure 16 is a plot of expected antenna pointing time ersus antenna gain for a set of assumed lunar parameters. From the above dis-ussion, it may be seen how a simple Earth sensing system on the LRV could greatly implify the antenna pointing problem.

A mechanically-steered directional antenna that will satisfy the above scheme s a parabolic reflector. This antenna would be mounted on a mast above the vehicle nd could be made of light weight material to decrease the inertial-movement prob-ems associated with mechanical steering systems. The beam could also be steered electronically through use of a phased-array antenna. However, simple arrays are ot capable of being steered over a hemisphere. Either a combination of electronic steering and mechanical steering could be used, or a more complex array could be leveloped. If the combination form were used, the array could be mechanically pointed in large steps and electronically pointed in fine steps. In this manner, it is ikely that the mechanical pointing would be required only occasionally. More com-plex arrays are available which could be steered over a hemisphere. However, the associated electronics consume a large amount of power and weight, and therefore this type of system should be considered as secondary to the mechanical steering or combination systems.

For the level of antenna gain and complexity indicated above, a parabolic reflec-tor which is mechanically steered is a logical choice.

One possible operating philosophy would be to transmit low data rate engineering telemetry while the vehicle is moving by using the low power transmitter and omni-antenna. When the rover stops, a portion of the power used for locomotion becomes available for the high power transmitter. The high gain antenna is now positioned, making available a high capacity channel for transmission of high data rate telemetry or television.

Since the average locomotion power is expected to be about 100 watts, a 20 watt high power transmitter appears to be a reasonable selection. Sizing of the antenna is now based on the results of the previous analysis for determining the expected time required for antenna pointing.

In order to optimize the traverse mode of the rover, it is desirable to minimize the total time required to take and transmit the television pictures used for navigation. The expected total time required for obtaining television pictures is shown graphically

in Fig. 16 as a function of antenna gain. The curve is the sum of the expected antenna pointing time and time required for transmission of five 600 line pictures. Assumptions include a 20 watt power amplifier and a number of typical lunar parameters.

Based on the results shown in Figure 16, a logical choice for a hig... g.... antenna would be one of at least 10 or 12 db gain. Since a small antenna involves less of a pointing problem, a 12 db gain antenna was chosen for this example. This could be achieved physically with a parabolic reflector of about one foot diameter.

7. Power subsystem. The LRV power subsystem is required to provide electrical power during both the flight and lunar traverse phases.

The standard transit operations power profile (Figure 17) shows that the peak loads are as much as eight times the cruise power loads. Since these peak loads are short in duration it is wasteful to meet them except by a storage device such as a battery. The LRV is therefore constrained to carry a battery as a portion of its power subsystem.

Traverse power requirements were estimated in the following manner. Landed vehicle earth weight is 480 lb. This is equivalent to 81.5 Lunar pounds or 20.4 lb. per wheel. The force transmitted to each axle is composed of three components. They result from the rolling friction at the wheels (R_r), soil deformation in and around the area of contact (R_m) and nonhorizontal terrain (R_s). While these components are influenced by such presently unknown factors as wheel design and terrain character, representative values have been determined empirically in studies with the SLRV. R_r is given as 7% of the weight at a wheel, R_m is $2R_r$, and R_s equals the weight on a wheel multiplied by the sine of the average vehicle slope. The latter is assumed to be 10°. Thus

$$R_r = 0.07(20.4) = 1.428 \text{ lb.}$$

$$R_m = 2(1.428) = 2.856 \text{ lb.}$$

$$R_s = 0.174(20.4) = 3.55 \text{ lb.}$$

The force per wheel is the sum or 7.8 lb./wheel. The electrical input power to each wheel, assuming a vehicle speed of 0.684 ft/sec. (1.25 km/hr.) and overall electrical/mechanical efficiency of 30% is:

$$7.8 \times \frac{746}{550} \times \frac{0.684}{0.3} = 24 \text{ watts/wheel.}$$

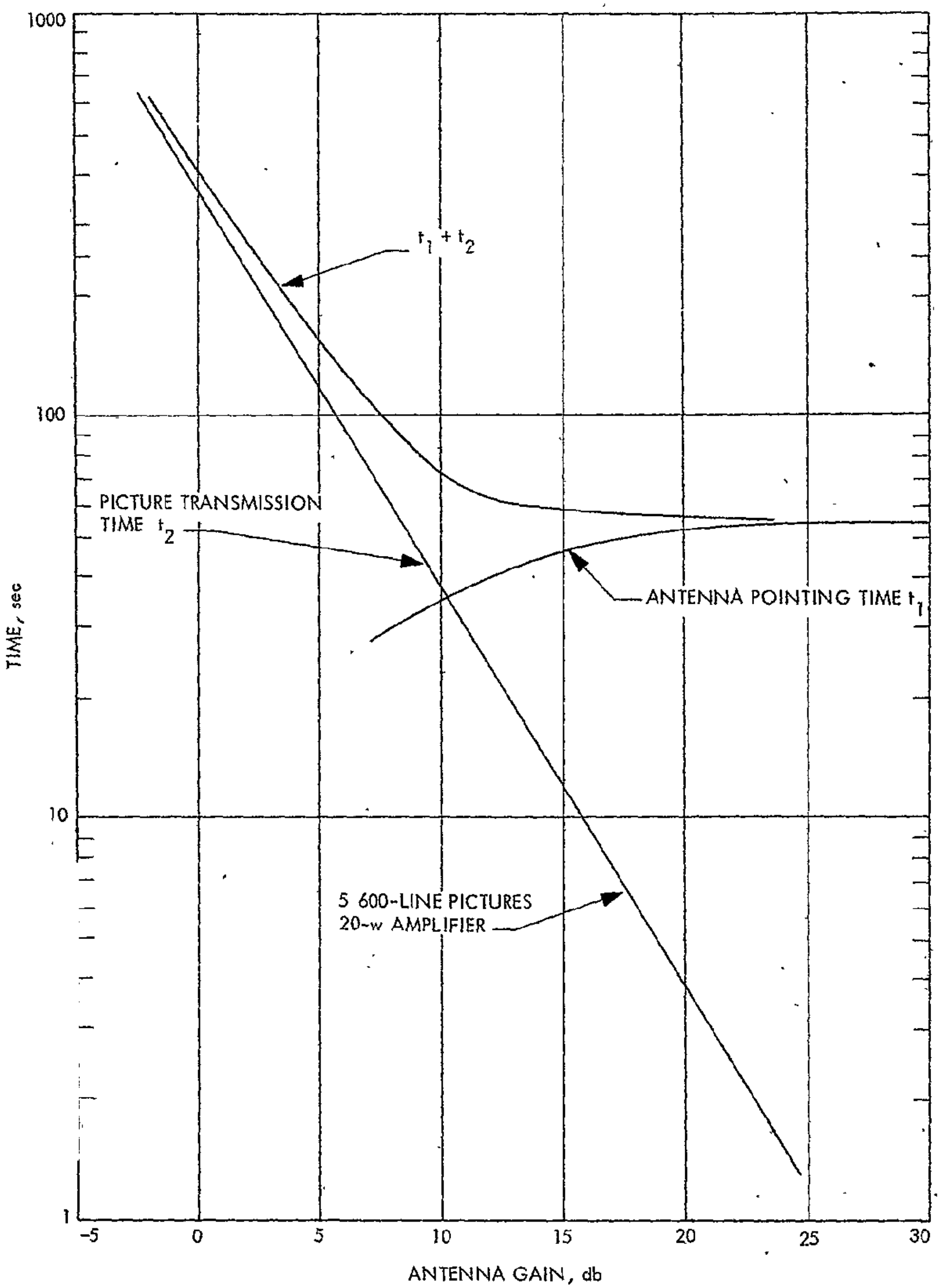

Fig. 16. Typical picture transmission time as a function of spacecraft antenna gain

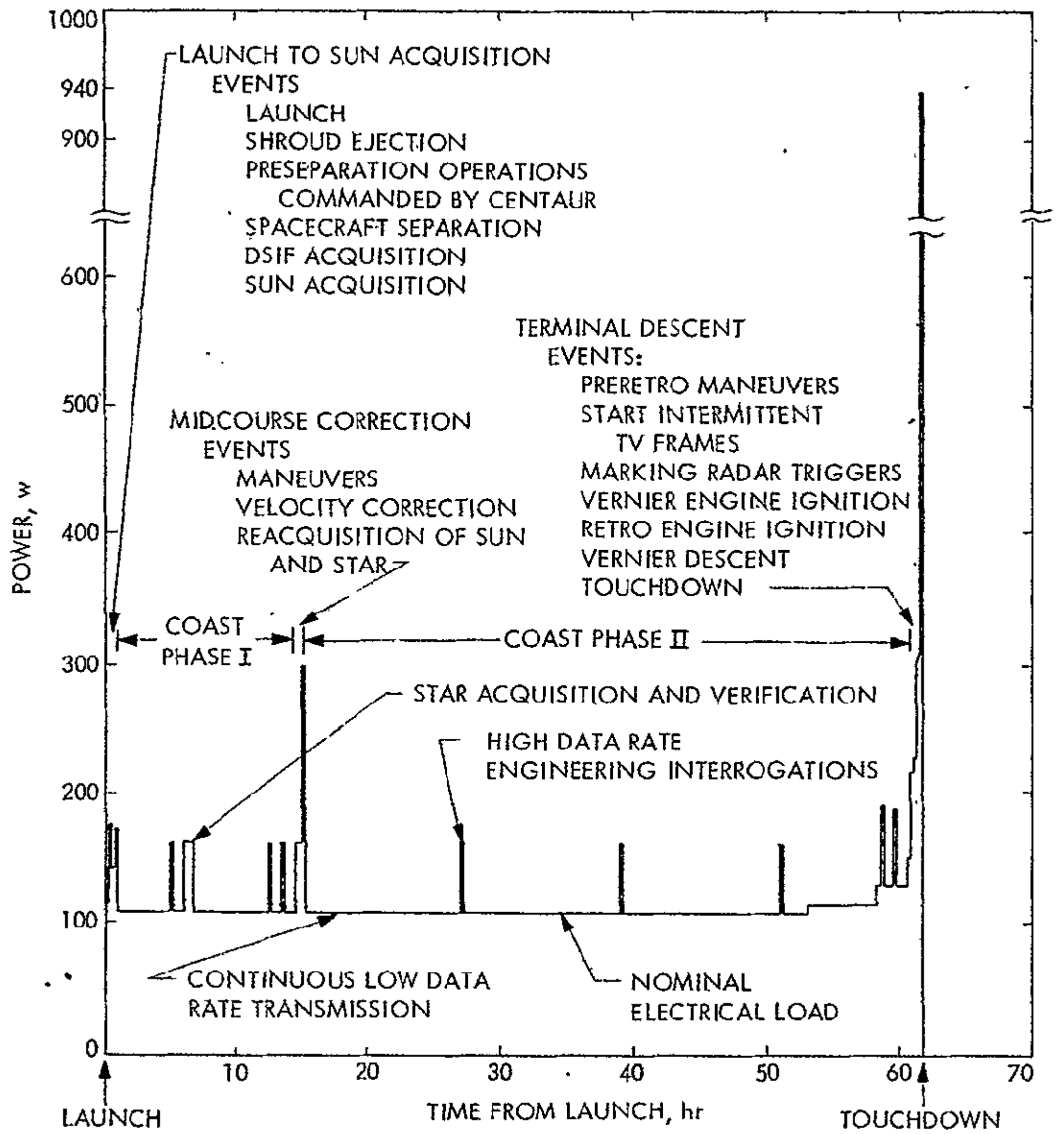

Fig. 17. Transit power profile

Total locomotion power is therefore 96 watts. Approximately 50 watts is the anticipated demand of the low-power transmitter, command receiver, obstacle and hazard detection electronics, control electronics, power conditioning equipment and battery charging. The total average power demand is therefore on the order of 150 watts, independent of whether the vehicle is moving or stationary during normal daytime operations. Power to the wheels in a stall condition is estimated to be 2-1/2 times the average locomotion power or 240 watts. Maximum locomotion power for vehicle movement will be that required to negotiate the maximum slope of 25°. Adding a 5° contingency for stones or other surface irregularities, the maximum $R_s = 10.2$ lb., the maximum force at each wheel is 14.5 lb. and the total maximum locomotion power is 180 watts.

Figure 18 shows a typical power profile during lunar traverse operations.

For the power subsystem, any of the three following configurations is feasible.

(1)	Solar Cells and battery	160 pounds
(2)	RTG and battery	145 pounds
(3)	Solar Cells, RTG and battery	130 pounds

Since the LRV is weight constrained, the third configuration is chosen for the baseline design.

For this arrangement it can be considered that a solar panel will supply half the average power, one RTG will supply the other half and the battery will supply peak loads.

Figure 19 shows weight vs. power for the RTG, fixed solar panel and pointable solar panel.

Peak power demand is during terminal maneuver when as much as 940 watts are drawn. During this time 300 watt hours must be provided by the RTG and Battery. The traverse demand is 150 watts, therefore the RTG provides half that or 75 watts. The total terminal time is one hour; thus the battery supplies 225 watt hours.

8. Surface drive and steering. Various concepts are being considered for driving the wheels including scaled up versions of the design studied in the SLRV program. This study investigated various types of mechanical as well as electrical driving systems. The electrical concepts appropriate for the requirements involved

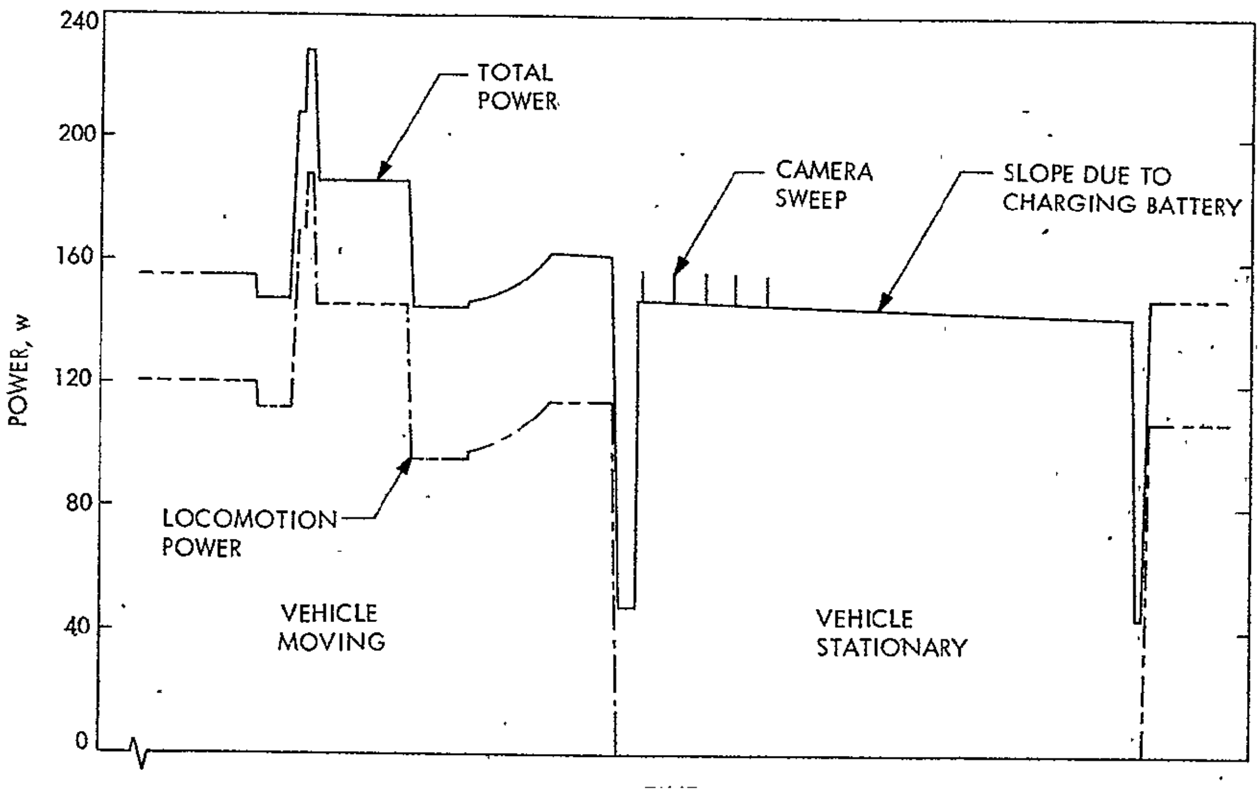

Fig. 18. Roving power profile

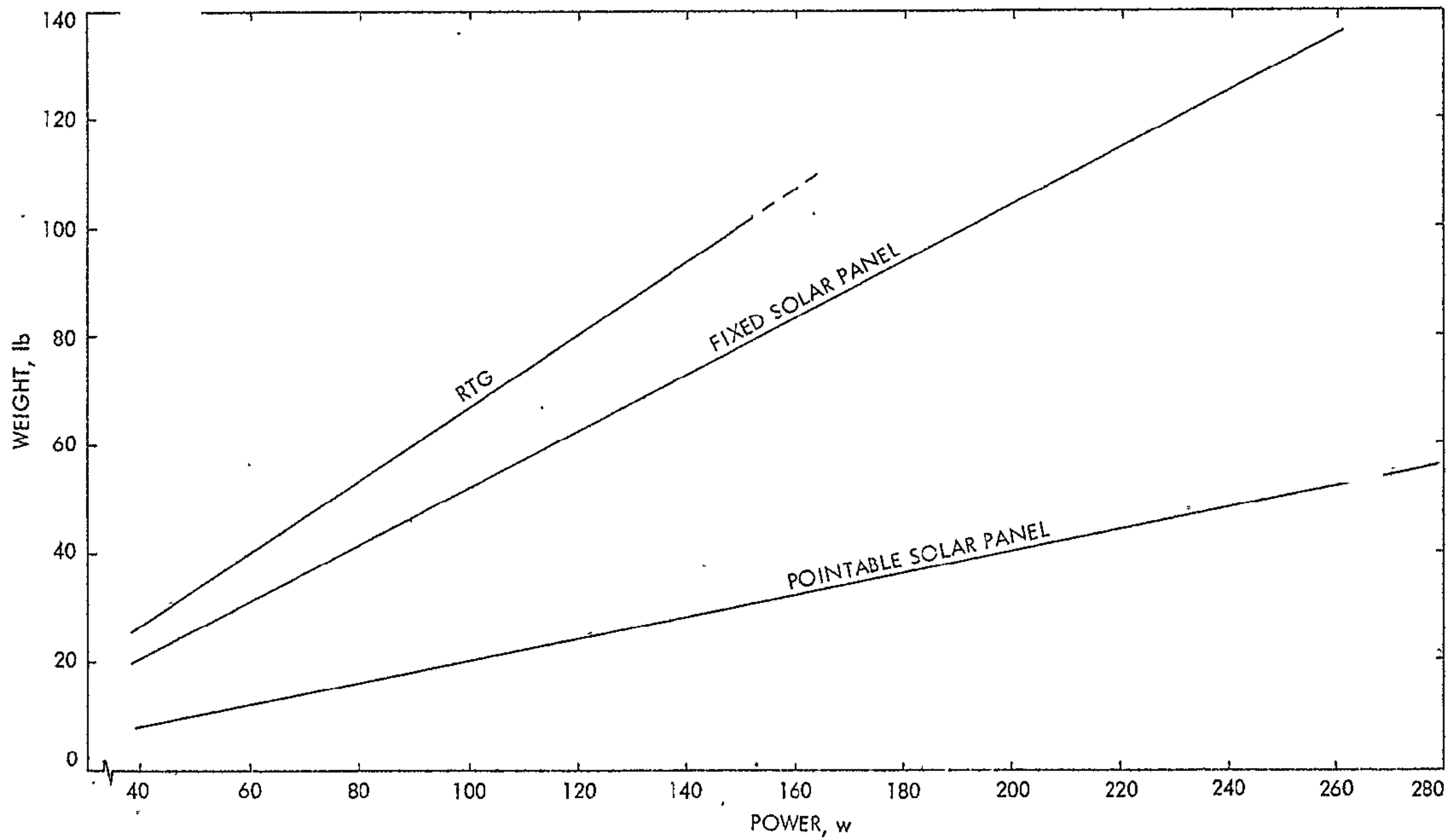

Fig. 19. Power supply weights vs power

AC and DC motors (standard and brushless) used in combination with: spur, planetary and "hunting tooth" (harmonic) type of gear reduction drives. The preferred design for the smaller SLRV vehicle was a sealed DC motor in combination with a planetary gearbox. This design incorporated a simple clutch design to permit individual free wheeling or drive of any of the six wheels. Besides providing for selective wheel driving and wheel release for motor failure, this provides the gearbox with protection during landing. Critical problems with this approach included the sealing problem for brush arcing and the heat dissipation problem which is aggravated by the wheel contact with the hot lunar surface.

A conventional Ackermann steering linkage with a 30° maximum steering angle is being considered for the baseline design. This configuration provides a wall-to-wall turning diameter of 30 feet. Both front and rear pairs of wheels are considered to be steered independently to permit normal leading wheel steering during forward or rearward travel. Steering is accomplished by electric actuators commanded from earth. Center point steering is provided to minimize the torque requirements of these actuators. The steering linkage also incorporates an extension link to permit wheel swiveling to the stowed position.

9. Structure. The major structural element of the roving vehicle is the central equipment compartment. It provides the body of the roving vehicle which contains the experiments and electronics in a thermally controlled environment. Attached to it are: the landing/mobility undercarriage, the descent systems "A" frame and the solar panel/high gain antenna mast. The structural concept of this octagonal body consists of sides of aluminum or magnesium alloy sheets which are tied together by a framework of corner angles, plus appropriate fittings as necessary for undercarriage and mast attachment. The concept of electronic installation considers the module design similar to the Mariner B approach with attachment to the inside of the box sides. The central portion of the compartment is available for experiments.

Integration of landing and roving functions into a single vehicle imposes relatively severe requirements on the design of structural and mechanical subsystems. Normal lunar surface roving operations will impose only nominal loads on the structure; however, provision must be made to absorb or dissipate the kinetic energy associated with the residual velocity of the vehicle at the instant of touchdown.

In establishing the baseline configuration, an attempt was made to minimize the landing loads on the structure by minimizing the impact while providing for large deflections on the undercarriage at touchdown.

However, for the configuration considered, the geometrical constraints (e.g. the Centaur shroud) limit the undercarriage deflections, and landing loads were found to constitute design loads.

Previous analytical and experimental investigations of mobility over soft soil and rough terrain have demonstrated desirable performance characteristics of relatively soft flexible wheels. Consideration of the lunar surface environment further suggested the use of all-metal wheels.

Several all-metal elastic wheel concepts were considered, including those developed for other lunar mobility applications (SLRV, Prospector, MOLAB, and LSSM). A wire mesh wheel concept was adopted for the baseline configuration since it appears to exhibit the desirable soft soil and irregular terrain mobility characteristics typical of flexible wheels. In addition, its ability to accommodate relatively large lateral deflections is advantageous in the landing process.

The investigation of wheel concepts was not exhaustive, and further study or test results may suggest modifications to this design, or the adoption of another flexible wheel concept.

In order to provide for landing impact absorption as well as for smooth riding characteristics on irregular terrain, three basic suspension system concepts were reviewed, namely: (1) swing axle; (2) single link trailing arm, and (3) four bar linkage. Evaluations of performance characteristics (roving as well as landing impact), weight, positioning for stowage in the shroud, ground clearance, and design simplicity led to the selection of the four bar linkage for the baseline configuration.

10. Temperature control. Since the thermal environment of the flight, descent, and landed conditions of this rover and the Surveyor are essentially the same, the gross temperature control concepts are considered to be similar as follows:

For the transit and descent, passive temperature control concepts are considered for propulsion and RADVS consisting essentially of insulating blankets and thermal coatings.

For roving, temperature control concepts of the central compartment are considered to be similar to that of the landed science and electronics compartment of the Surveyor. The bottom and sides of this compartment are thermally insulated using external thermal coatings with insulating mylar inside these surfaces. Temperature control is then maintained by thermal switches contacting the special reflective glass surface of the compartment top or sides. Special precautions will be necessary to thermally insulate the wheels which are in contact with the hot (or cold) lunar surface. It should be pointed out that the thermal problem of the wheels and their drive motors is one of the critical new areas requiring further study. Overheating problems have already been encountered in preliminary wheel drive testing. Another major problem area concerns the repeated nighttime survival requirement for the one year lifetime mission duration. The heating problem for this condition may enhance the value of a nuclear source for the rover power system or may introduce the need for local isotope heating pads.

11. LRV Scientific payloads. Table 4 shows a sample list of instruments which could be carried by the LRV in various combinations as permitted by the weight and power limitations of the LRV system and its operating modes.

a. Photo-imaging. The objectives of the photo-imaging instrument are as follows:

(1) Provide real-time images that can be used to guide the roving vehicle;

(2) Acquire dimensionally stable and photometrically faithful images from which topographic maps can be made by photogrammetric methods;

(3) Provide reconnaissance eye-type geological information;

(4) Provide near-field information on surface structure and texture, with th capability to detect particle sizes down to at least 1 mm.

A variety of sensors and camera systems could be adapted to the roving mission; however, the selected system should meet the following requirements esse tial to the objectives of the roving vehicle mission.

(1) A stereographic baseline of the camera system of preferably 3 feet but no less than one foot. The baseline may be vertical or horizontal.

Table 4. Example instrument payload for geological reconnaissance

Instrument/Experiment	Weight in Lbs.	Objectives
1. Photo-Imaging	8	Provide images for guidance and displaying the surface topographic form, structure, stratification and texture. Provide high resolution images of acquired sample material.
2. Hard-Rock Drill & Particulate Sampler	17	Acquire samples.
3. Sample Processor/ Transporter	5	Prepare sample from drill for the fixed-analysis instruments.
4. X-ray Diffractometer	20	Identify the types and relative abundance of crystalline phases present in lunar samples.
5. X-ray Spectrometer		Provide information on the elemental composition of lunar material.
6. Petrographic Microscope	15	Identify lunar rock types and determine their genesis -- detection and composition of glass -- examination of particulate material as to texture, etc.
7. Gas Chromatograph	10	Identify some of the volatile constituents in lunar surface material.
8. Gamma-ray Spectrometer	14	Identify characteristic gamma-ray lines for radiogenic isotopes of potassium, uranium, thorium.

(2) At each position of the roving vehicle from which an image is obtained, a measurement should be transmitted that gives the orientation with respect to the local vertical within 0.5 degree.

(3) The resolution of the imaging system should be at least 6 minutes of arc at 20-percent sine wave response.

(4) The imaging system should be able to scan in elevation at least 40 deg above and 60 deg below the horizontal axis of the LRV.

(5) The system should provide images with at least a 36-decibel peak-to-peak signal-to-rms noise ratio, for scene highlight luminences ranging from 25 foot lamberts to 2500 foot lamberts; also, it is desirable that the system have a 24-decibel peak-to-peak signal-to-rms noise ratio for operating in lunar shadows where the scene highlight luminences may range down to a few foot-lamberts.

(6) The system should provide for 360 deg panoramic coverage or coverage of a select sector.

(7) The imaging system should be able to distinguish at least 13 grey levels.

(8) The total system weight including spacecraft attachments will be about 8 pounds, and it can be packaged into about 0.4 cubic ft.

b. Hard Rock Drill and Particulate Sampler. It is desirable that a hard-rock drill technique be used for sample acquisition on this mission. This is because of the probability that the Rover will find hard-ground outcrops, flows, or perhaps large pieces of consolidated ejecta, and it would be very desirable to analyze the mineralogy and petrology of this type of material.

Under development is a small, low-powered rotary impact sampling drill which will produce the fragmented rock powder required by the analytical instruments from rock as hard as dense basalt. Since the drill is not an efficient sampling device in highly vesicular rock and some underdense particulates, attempts willl be made to incorporate into this drill a particulate acquisition device. Should this not prove feasible, two separate samplers would be used. In hard rock the drill would require about 20 minutes to produce samples for all the instruments, use about 100 watts peak power, have a depth capability of about a foot, and weigh approximately 10 pounds. The particulate sampler would acquire and transport samples in less than five

minutes, draw about 25 watts of power, and weigh approximately 7 pounds. Both devices could be packaged into a volume of 1.5 x 0.5 x 0.5 ft.

c. Geosampler/Transporter. This instrument consists of a small, very lightweight sampling device, suitable for lunar surface and near-subsurface sampling conditions, and a simple transporter arrangement for presenting sample material to an array of fixed analysis scientific instruments. Thus, the geosampler/transporter will perform two basic functions:

(a) Sample acquisition from a fixed point under the roving vehicle

(b) Sample distribution from the various sample devices to an analysis point.

A study of the soil properties of the lunar surface as portrayed by the Surveyor and the Luna landing dynamics and experiments suggests a granular surface with a bearing strength of perhaps 5 psi that can be perforated without a great deal of energy and hard drilling.

A number of lightweight devices would be applicable for this purpose especially if hard-rock piercement could be handled by another separate device.

For example, a so-called rigid helical conveyor, with drill tip, could satisfy the requirements. The instrument would weigh less than 5 pounds including deployment mechanism and sample chute and tray. It would be capable of sampling the "typical" lunar soil to a depth of perhaps 5 inches. It size-sorts particles so as to diminish the content of those over 500μ and largely reject those over 1000 μ; below 500 μ it does very little sorting. The instrument has drilling capability in weakly consolidated material and pumice-like rock. The device requires access to the hard rock drill and particulate sampler and also to the fixed-analysis instruments. It can be packaged into about 0.5 cubic ft.

d. Combination X-Ray Diffractometer and Spectrometer. The X-ray diffractometer will be used to conduct mineralogical analyses of lunar surface material acquired at a number of fixed points on a roving vehicle traverse. The primary objective of this instrument is to identify the types and relative abundances of the various crystalline phases expected to be present in a lunar sample. The instrument will provide diffraction data of sufficient quality to identify any of the major rock-forming and accessory minerals.

The X-ray spectrometer will be used to conduct an elemental analysis of lunar surface material acquired at a number of fixed points on a roving vehicle

traverse. This mode of analysis can detect elements from sodium through uranium; however, only those elements from sodium through nickel are expected to be present in sufficient quantity to allow detection. In general, the instrument should be capable of detecting elements with the following limits of sensitivity (by weight):

Element	Limits of Sensitivity %
Sodium	0.5
Magnesium	0.2
Aluminum	0.1
Silicon	0.1
Sulfur	0.1
Chlorine	0.1
Potassium	0.05
Calcium	0.05
Titanium	0.05
Chromium	0.02
Manganese	0.02
Iron	0.02
Nickel	0.02

The combination of these two instruments represents a powerful tool for understanding lunar geology. It should be used in conjunction with imaging system, which should be capable of detecting the textural properties of lunar material to a resolution of at least 1 mm.

This combination experiment will weigh about 20 pounds and as a fixed-analysis instrument it would receive its samples from the hard rock drill and particulate sampler and also the geosampler/transporter. The complete experiment can be packaged into about 0.6 cubic ft, and in operating mode, it will require about 5 watts of power. It should be physically located on the LRV near the geosampler/transporter. Also, lunar surface sample material should be viewed by the imaging system either before or after analysis.

e. Petrographic Microscope. The petrographic microscope is designed for remote observation of crushed rock samples in transmitted light. The objectives are

Identification of lunar rock type and genesis thereof, by:

identification of rock texture .

identification of shapes and relative sizes of different mineral grains

determination of relative abundances of minerals in sample

detection of glass and estimate of composition of glass by refractive index

determination of particle size and shape distribution of particulate surficial material

identification of phases present in small amounts

The petrographic microscope system was conceived as an experiment of low weight and power which could provide textural information on rocks and particulate material from the lunar surface. The type of data provided by this system would preclude ambiguities in the interpretation of other petrological experiments. The objectives listed above are of considerable impoertance in the scientific exploration of the Moon. Except for determination of bulk mineralogy, and analysis of returned samples, these data will not be forthcoming from other instruments.

In operation, a sample of lunar material consisting of sized particles in the range of 10 to 300μ is delivered to the microscope from the spacecraft sample processor/distributor system. The microscope system separates these particles into fine-grained and coarse-grained fractions. Each fraction is then immersed in a clear isotropic medium of known refractive index. The particles form a mono-particle layer with their tops in a plane. The sample is transported to the field of view of a lens system which displays its magnified image onto the optical sensor of a television system. The immersed sample moves in steps across the field of view of the objective lens. Several images individually focused at different design depths in the particle layer are obtained for each field of view. The operation sequence during viewing involves:

(1) Movement of some immersed particles into field of view;

(2) Individual images at several planes of focus taken;

(3) Movement of more immersed particles into field of view and imaging sequence repeated.

Every particle is viewed in both plane-polarized and cross-polarized light. The samples are stored and are available for recovery and review.

The imaging subsystem consists of:

(1) A light source capable of narrow-bandwidth output, and a light condenser;

(2) An objective lens to project a magnified image to a television camera;

(3) Polarizing filters which will allow particles to be viewed in both plane-polarized and cross-polarized light;

(4) A television camera capable of recognizing 10μ particles at any spot over a field of view of 0.5 x 0.5 mm.

The total system weight including television camera is approximately 15 to 18 pounds. It can be packaged into about 0.5 cubic ft and it would require about 3 watts of power. This is one of the fixed analysis experiments. Therefore, it is operated in conjunction with the hard rock drill and particulate sampler and also the geosampler/transporter.

f. Gas Chromatograph. The objective of the gas chromatograph experiment is to provide an analysis of the volatile constituents in lunar surface material. It should detect the following compounds and elements to a specified resolution even when their quantity is as low as 10^{-10} mole of sample gas within the ovên structure.

1. Hydrogen (H_2)
2. Oxygen (O_2)
3. Nitrogen (N_2)
4. Carbon Monoxide (CO)
5. Water vapor (H_2O)
6. Methane (CH_4)
7. Ethane (C_2H_6)
8. Propane (C_3H_8)
9. Butane (C_4H_{10})
10. Methanol (CH_3OH)
11. Ethanol (C_2H_5OH)
12. Propanol (C_3H_7OH)
13. Formic Acid (HC OOH)
14. Acetic Acid (CH_3COOH)
15. Butyric Acid (C_3H_7COOH)
16. Formaldehyde (HCHO)
17. Acetaldehyde (CH_3CHO)
18. Propionaldehyde (C_2H_5CHO)
19. Acetone (CH_3COCH_3)
20. Acetonitrile (CH_3CN)
21. Benzene (C_6H_6)
22. Toluene ($C_6H_5CH_3$)
23. Ammonia (NH_3)
24. Acrolein (CH_2=CHCHO)
25. Acetylene (CH = CH)
26. Carbon Dioxide (CO_2)
27. Hydrocyanic Acid (HCN)
28. Hydrogen Sulfide (H_2S)

In operation, a sample of lunar crustal material is collected by the appropriate sampling device on the LRV and placed in the sample processor/

distributor for delivery to the gas chromatograph. The delivered sample is passed through a funnel in the top of the chromatograph into a small oven. The oven is then sealed by a pneumatically operated mechanism and heated to release gaseous material that may be present in the sample. The gases thus liberated are injected into a helium carrier gas stream in the form of a tight slug. The sample gas is then divided and swept through analytical packed columns. The constituents of the sample gas will have more or less affinity for the packing material in the columns. Through the mechanisms of absorption and/or chemical equilibrium, the passage of each constituent through the columns is impeded for a distinct, reproducible time interval unique for each unknown. This retention time is measured at the effluent end of each column by a signal from a detector which senses the presence of any unknown material other than the helium carrier gas. The outputs from the detectors are fed into the LRV for data processing and transmission back to Earth. From this transmitted data, the identity and approximate quantity of each unknown volatile constitutent in the sample can be determined.

This is one of the fixed analysis experiments. Therefore, it is operated in conjunction with the hard rock drill and particulate sampler and also the geosampler transporter. As long as the individual particles have dimensions of less than about 5 mm, sample grain size is not important. Oven temperatures can be selected to one of three temperatures: 150, 325, 500° C $\pm$ 10°C. The instrument is capable of cycling through analyses of a large number of selected samples; however, a maximum of thirty minutes should be allowed for each complete cycle. The instrument weighs about 10 pounds.

g. Gamma-ray Spectrometer. The gamma-ray detector will be a scintillation crystal of sodium iodide coupled to a photomultiplier tube. A shell of plastic scintillator will surround the inorganic crystal for the purpose of eliminating an unwanted response to the charged particle flux. After suitable shaping and amplification, the analog output signal from the photomultiplier is digitized and stored in a memory. The contents of this memory will be read out periodically and transmitted back to the spacecraft for relay to Earth. Much of the electronics can be time-shared with a nondispersive X-ray spectrometer, if the mission profile permits. Allowing for a 3 x 3-inch crystal of NaI, the total weight of the instrument will be 12 to 14 lb and 6 to 7 lb if nondispersive spectroscopy is included on a time-sharing basis. Power requirements will be about 4 watts for a complete instrument and 1 to 2 watts for the detector

and its associated electronics. The gamma-ray detector should be deployed away from the main mass of the roving vehicle at a height of 0.5 to 2 feet above the surface. The vehicle must be clean of interfering sources of radioactivity. An RTG power supply will make the experiment impossible.

REFERENCES

1. Lunar Exploration Program Study. Working Papers Contributed by JPL to NASA Lunar Exploration Program Review at GSFC (Jan., 1967) Pasadena, California.
2. Space Research, Directions for the Future. (1965) Space Science Board, National Academy of Sciences, National Research Council, Washington, D. C.
3. The Space Program in the Post-Apollo Period. A Report of the President's Science Advisory Committee (February, 1967) Washington, D. C.
4. A Strategy for Scientific Exploration of the Terrestrial Planets, General Technical Document 900-28, (July 1967) JPL, Pasadena, California.
5. Scientific Objectives and Requirements of a Lunar Orbiter to Investigate the Field and Particle Environment of the Moon. Advanced Studies Document 760-8, (July 1967) JPL, Pasadena, California.
6. Scientific Instruments for Lunar Exploration, Part A, Lunar Orbiter (Unmanned) (January 15, 1967) Engineering Planning Document 472, JPL, Pasadena, California.
7. Scientific Instruments for Lunar Exploration, Part B, Surveyors, Roving Vehicles, and Rough-Landed Probes (May 1, 1967) Advanced Studies Document 760-3, JPL, Pasadena, California.
8. Prospects for a Lunar Field Assistant, Advanced Studies Document 760-7, (July 1967) JPL, Pasadena, California.